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USATECOM PROJECT NO. 4-3-0250-11/12/13
PART TWO OF TWO PARTS

REPORT
OF THE ENGINEERING FLIGHT TEST
PERFORMANCE PHASE
OF THE
OH-4A HELICOPTER, UNARMED (CLEAN) and ARMED
WITH THE XM-7 or XM-8 WEAPON SUBSYSTEM

AUGUST 1964

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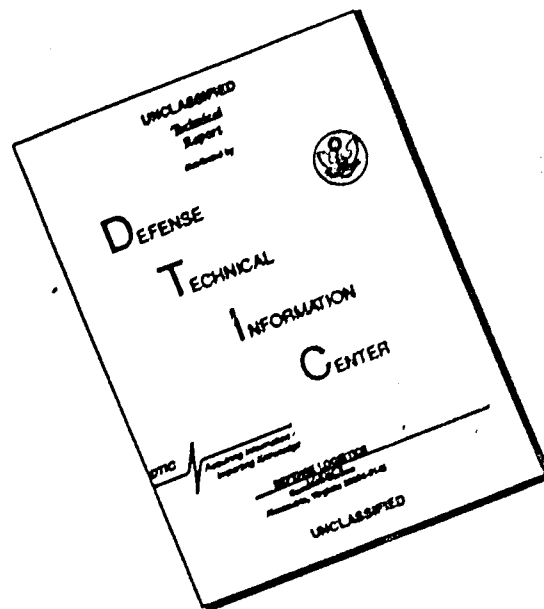
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U. S. ARMY AVIATION TEST ACTIVITY

① PART TWO OF TWO PARTS
REPORT
OF THE ENGINEERING FLIGHT TEST
--PERFORMANCE PHASE--
OF THE
OH-4A HELICOPTER, UNARMED (CLEAN) AND ARMED
WITH THE XM-7 OR XM-8 WEAPON SUBSYSTEM. Part II.

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FOREWORD

Essential to an understanding of the results of aircraft testing is an understanding of the differences between engineering and service testing.

Engineering testing, using instrumented aircraft and calibrated instruments, can determine and record the exact performance, control response and limits, engine performance and power available, through accurate measurements and reduction of data to standard conditions. Thus, it is possible to determine when an aircraft is approaching or exceeding design limits or other specified criteria.

Service testing, using aircraft in standard configuration, results in a qualitative evaluation for user-type information. This information is based on a broad scope of pilot experience and technique provided by pilots ranging from those recently out of school to those with considerable field operational experience. The installed instruments and gauges are used to determine significant operating data. These instruments are not usually calibrated but represent typical instruments found in production helicopters. These instruments and gauges are verified for accuracy within acceptable tolerances but do not attain the precision provided by the calibrated equipment used for engineering testing.

The service test-pilot makes qualitative observations on only what he experiences during normal service flying. These observations are not correlated to such factors as the margin of control remaining or exact rates of control response. Exact measurements of such factors are necessarily the responsibility of the engineering test agency. Thus, service testing may show that the aircraft is suitable for performing a mission when, actually, flight has been performed close to, or within, control margins specified by military specifications. What may appear to be discrepancies between service and engineering test reports is actually the difference between qualitative and quantitative reporting.

The Light Observation Helicopter evaluation is the first combined aircraft engineering and service test program that has resulted in coordination of reports and comparison of reports prior to procurement decision. Caution must be exercised, therefore, to preclude taking an item out of context in any one report to establish a particular position. Seeming inconsistencies can be reconciled only by examination of all reports with due regard to the specific conditions under which the test was accomplished.

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ABSTRACT

Performance flight tests were conducted on the OH-4A to determine the compliance with the Light Observation Helicopter (LOH) Military Characteristics and to check the contractual guarantees outlined in the OH-4A Model Specification.

The U. S. Army Aviation Test Activity (USAATA), Edwards Air Force Base, California, was designated Executive Test Agency for the confirmatory engineering tests in the LOH Program and is responsible for test execution and test reporting of its assigned phase.

Engineering flight tests were conducted by the U. S. Army Aviation Test Activity at Edwards Air Force Base, California, and at auxiliary test sites at Bakersfield and Bishop, California. A total of 121 flights were conducted for 73:45 productive flight hours. These tests were accomplished during the period of 13 March to 30 June 1964.

All Model Specification performance guarantees and requirements stated in the LOH Military Characteristics were evaluated using power available and fuel flow characteristics as defined in Engine Model Specification No. 580-A.

The OH-4A met the contractual performance guarantee for maximum airspeed.

The OH-4A did not meet the contractual performance guarantees for endurance or for 35 degrees centigrade (C) out-of-ground effect (OGE) hovering ceiling.

The OH-4A met the Military Characteristic requirement for 3 hours endurance with a 400 pound payload.

The OH-4A did not meet the Military Characteristics requirements for cruise speed, endurance at 85 percent cruise power, OGE hot day hover performance at 6000 feet or OGE hover performance at overload gross weight, sea level standard day.

The OH-4A performance was summarized using power available as defined by Engine Model Specification No. 580-E.

Takeoff performance was satisfactory for all conditions that allowed a hovering skid height of 2 feet or more. Hovering performance was satisfactory at low altitude standard day conditions but deteriorated rapidly with increasing ambient temperature and increasing altitude. Climb and autorotation performance was satisfactory for a helicopter of this

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power and weight class. Blade stall was not encountered. Vibration levels were exceptionally low.

This program was essentially an airframe evaluation, but a section has been incorporated into Appendix II in order to describe pertinent characteristics of the T63-A-5 engine.



PHOTO 1 - OH-4A



PHOTO 2 - OH-4A

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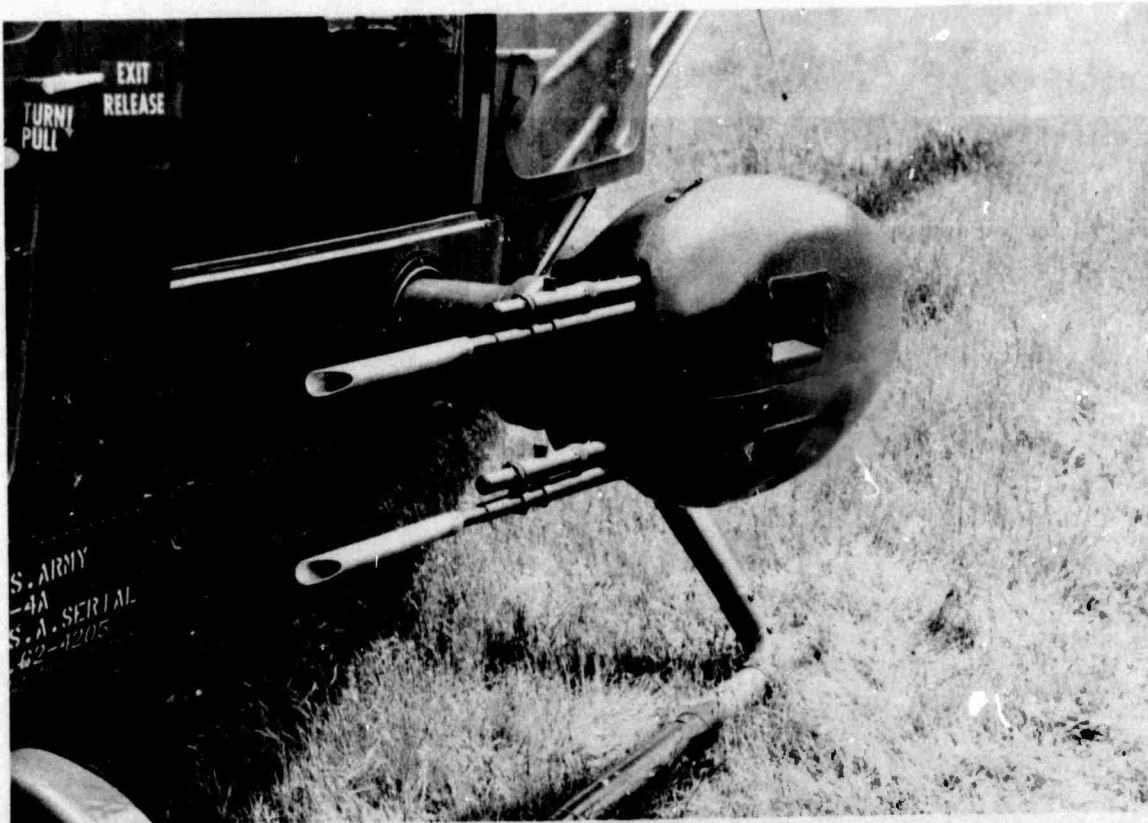


PHOTO 3 - XM-7 ARMAMENT KIT

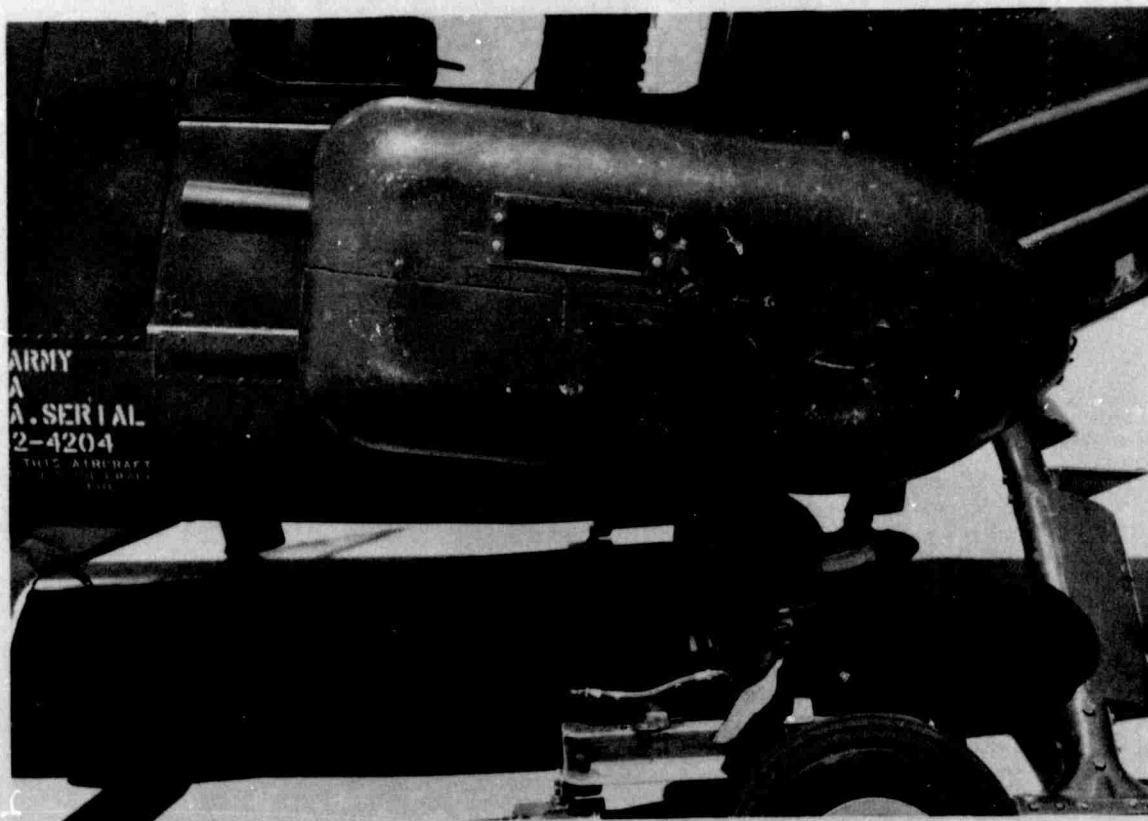


PHOTO 4 - XM-8 ARMAMENT KIT

SECTION 1 - GENERAL

1.1 REFERENCES

See Part I, Section 1, References a. to j. inclusive, and the following additional references:

k. Preliminary Report, "Flight Evaluation of the T-63-A-5 Gas Turbine Engine Installed in the UH-13R Helicopter," USATECOM Project No. 4-4-0240-02, U. S. Army Aviation Test Activity, January 1964.

l. Model Specification, Light Observation Helicopter, OH-4, 15 September 1961; Revision No. 1, 8 December 1961.

m. Model Specification No. 580-A, Model T63-A Engine, 12 December 1960.

n. Model Specification No. 580-E, Model T63-A-5 Engine, 24 June 1963.

o. Letter, SMOSM-PAIA-2, Headquarters, U. S. Army Aviation Materiel Command, 4 April 1964, subject: "Compliance Check of Manufacturer's Guaranteed Performance and Competitive Performance Evaluation."

p. Final Report of "OH-13H Gross Weight Increase/XM-1 Armament Kit Performance Test," U. S. Army Aviation Test Activity, June 1963.

1.2 AUTHORITY

See Part I, Section 1.

1.3 OBJECTIVES

The objective of this program was to conduct engineering performance flight tests of the Light Observation Helicopter (LOH) Prototype OH-4A to (a) confirm contractor compliance with the approved Army Military Characteristics for an unarmed (clean) and armed OH-4A helicopter; (b) provide data to assist in selecting an LOH design for possible future production; and (c) determine if the contractor performance guarantees were met.

1.4 RESPONSIBILITIES

See Part I, Section 1.

1.5 DESCRIPTION OF MATERIEL

See Part I, Section 1; Part I, Section 3, Appendix 11; and Part II, Section 3, Appendix 11.

1.6 BACKGROUND

See Part I, Section 1.

1.7 FINDINGS

The takeoff performance of the OH-4A was satisfactory for all gross weights and ambient atmospheric conditions that allowed a hovering skid height of 2 feet. Under conditions that allowed a hovering skid height of approximately 2 to 2-1/2 feet, the minimum distance to clear a 50 foot obstacle was produced by a "level acceleration from a 2 foot hover" takeoff technique. Under conditions that allowed a hovering skid height greater than 2-1/2 feet, shorter takeoff distances were obtained using a simultaneous "climb and accelerate from light on skids" technique. Greater pilot proficiency was required to execute the "climb and accelerate" technique with consistent results. Engine rpm governing deficiencies required that the pilot monitor engine operation closely during takeoffs while using either technique.

Hovering performance was adequate based on power data obtained from the T63-A-5 Model Specification 580-E at lower altitudes on a standard day. Performance deteriorated rapidly with increasing ambient temperature and altitude. Based on power data from the T63-A Model Specification 580-A, the hot day (+35 degrees centigrade (C)), normal gross weight hovering performance guarantee was not met. The out-of-ground effect (OGE) hovering ceiling guaranteed was 4300 feet + 10 percent. The actual OGE hovering ceiling at the gross weight and ambient conditions specified was 825 feet pressure altitude. The Military Characteristics, Light Observation Aircraft (See Part I, Section 1, paragraph 1.1a) required an OGE hovering ceiling of 6000 feet pressure altitude on a 35 degree C day at normal gross weight (2573 pounds). The maximum gross weight at which the OH-4A could hover OGE at these ambient conditions was 2085 pounds. The Military Characteristics also required OGE hover at sea level on a standard day at overload gross weight (2900 pounds). The maximum gross weight at which the OH-4A would hover OGE at sea level on a standard day was 2730 pounds.

The performance guarantees and Military Characteristics requirements were based on power available as defined in Engine Model Specification 580-A. The hovering performance summaries were based on Engine Model Specification 580-E. This revised model specification results

in better standard day performance below critical altitude due to a higher torque limit but more rapid deterioration of power available with increasing ambient temperature. The overload gross weight (2900 pounds) OGE hovering ceiling was 2200 feet on a standard day. The maximum gross weight for hovering OGE on a 35 degree C day at 6000 feet was 1986 pounds. The OH-4A would not hover OGE at sea level on a 35 degree C day at normal gross weight (2573 pounds).

The climb performance of the OH-4A was satisfactory. With a climb start gross weight of 2573 pounds at sea level, the service ceiling was 20,150 feet on a standard day. Sea level rate of climb was 1100 feet per minute at maximum continuous power and 1550 feet per minute at takeoff power. With a climb start gross weight of 2900 pounds at sea level, the service ceiling was 15,600 feet on a standard day. Sea level rate of climb was 810 feet per minute at maximum continuous power and 1250 feet per minute at takeoff power.

During level flight, the maximum airspeed performance guarantee was met. The maximum airspeed guaranteed was 100 knots \pm 10 percent at normal gross weight (2573 pounds) on a sea level standard day at maximum continuous power. Based on Engine Model Specification 580-A, the maximum true airspeed (TAS) was 96.5 knots at these conditions. The Military Characteristics, Light Observation Aircraft, required 110 knots at the same conditions. This requirement was not met by a margin of approximately 12.3 percent. The guaranteed endurance mission of 3.0 hours \pm 10 percent was not met by a margin of approximately 0.37 hours (13.6 percent). The Military Characteristics required a 400 pound payload to be carried for 3 hours. At the airspeed for minimum power required in level flight, this requirement was met by a margin of approximately 0.18 hours (6 percent). The Military Characteristics also required 3 hours endurance at 85 percent cruise power. This requirement was not met by a margin of approximately 0.45 hours (15 percent).

The equivalent flat plate area of the XM-7 armament system was found to be 2.46 square feet and that of the XM-8 was 1.57 square feet. This reduced the sea level standard day at normal gross weight maximum speed from 111.5 KTAS to 105 KTAS with the XM-7 and to 107.5 KTAS with the XM-8 at takeoff power available as defined by Model Specification No. 580-E. Center of gravity (C.G.) did not significantly affect level flight performance. Maximum airspeed was generally limited by power available at lower altitudes and by the "never exceed" (Vne) airspeed from the Federal Aviation Agency Type Inspection Authorization (FAA TIA) (See Part I, Section 1, paragraph 1.1.g) at higher altitudes.

Vibration levels were exceptionally low and blade stall or rotor compressibility effects were not encountered.

While changing collective pitch control settings, the pilot had to monitor rotor speed and use the "beep" switch to maintain a desired rpm.

Autorotational descent performance was satisfactory and typical for a helicopter the size and weight of the OH-4A. The minimum rate of descent was 1495 feet per minute. This was obtained at 48.0 knots calibrated airspeed (KCAS) and the minimum power-off rotor speed of 374 rpm.

1.8 CONCLUSIONS

None.

1.9 RECOMMENDATIONS

None.

SECTION 2 - DETAILS AND RESULTS OF SUB-TESTS

2.0 INTRODUCTION

Performance flight tests on the OH-4A Light Observation Helicopter were conducted by the U. S. Army Aviation Test Activity at Edwards Air Force Base, California. Sea level and high altitude testing was accomplished at Bakersfield and Bishop, California, where altitudes from sea level to 9500 feet and a wide range of ambient temperatures were available. A total of 121 flights were conducted for 73:45 productive flight hours. The tests were accomplished during the calendar period of 13 March 1964 to 1 July 1964.

Level flight performance data were obtained at Edwards Air Force Base, and at the sea level site at Bakersfield, California. Climb performance data were obtained entirely at Bakersfield where the climbs could be initiated near sea level. Autorotational descent performance data were collected during descents after completing other tests. Takeoff tests were accomplished in the Bishop, California area where pressure altitudes of 4100 feet (Bishop Airport) and 9500 feet (Coyote Flats) were available. Hovering performance data were collected at all of the test sites.

All tests were conducted in stabilized non-turbulent air so that accurate performance data could be obtained. The test data were recorded by hand from sensitive instruments or automatically using a photo panel.

Power available and fuel flow as specified in the T63-A Engine Model Specification 580-A were used to check the contractual guarantees and the desired performance as defined in "Military Characteristics, Light Observation Aircraft" (See Part I, Section 1, paragraph 1.1a) for hovering, maximum airspeed and endurance.

Summary performance was obtained using power available and fuel flow information based on the T63-A-5 engine referred characteristics as defined by model specification 580-E on a sea level standard day. Engine performance at the 6000 foot altitude, 95 degree Fahrenheit (F) ambient temperature and all other altitude temperature combinations were obtained from curves of standard power deterioration with increasing altitude and temperature. These curves were derived using standard engineering methods, and were verified with test stand engine calibration data on five different T63-A-5 engines.

Using the power available as described above, it was determined that a T63-A-5 engine, which will produce the minimum 580-E specification sea level standard day power, will not produce the 206 shaft horsepower (SHP) at 6000 feet, 95 degrees F as shown in the 580-E model specification. Such an engine will, in fact, only produce 199 SHP under this altitude, ambient temperature condition (See Section 3, Appendix II).

2.1 TAKEOFFS

2.1.1 OBJECTIVE

Takeoff tests were conducted to determine the performance of the OH-4A helicopter under conditions in which a vertical takeoff could not be made to clear a 50 foot obstacle. Under these conditions, a short acceleration close to the ground will enable the aircraft to operate out of relatively short fields.

2.1.2 METHOD

Takeoff tests were conducted to obtain curves of climb-out airspeeds versus distance required to clear 50 feet. Each curve was obtained by conducting a series of takeoffs using various climb-out airspeeds. During each series, ballast was added or removed as necessary so as to maintain the desired excess power available conditions as fuel was consumed and ambient temperature varied.

These tests were accomplished at two altitudes and several gross weights to define the performance over as wide an envelope as possible. Takeoff performance was evaluated using two takeoff techniques which are commonly used in the field. A ground operated Fairchild Flight Analyzer was used to produce a photographic record of time, horizontal distance and vertical distance for each takeoff.

The climb-out airspeed range used for each series of takeoffs varied from the minimum achievable to maximum practical airspeed (approximately 50 knots indicated airspeed (KIAS)). All takeoffs were performed in winds of 3 knots or less. These tests were conducted at density altitudes of approximately 5000 to 10,000 feet. Gross weight was varied from 2250 to 2930 pounds at the mid center-of-gravity (C.G.) location (Station 102.0). All takeoff tests were conducted with a main rotor speed of 394 rpm (100 percent power turbine speed (N₂)). using takeoff power.

2.1.3 RESULTS

The results of the takeoff performance tests are graphically presented in Figures No. 1 through 17, Section 3, Appendix I.

2.1.4 ANALYSIS

a. Comparative Techniques

Two takeoff techniques were used during this evaluation. Each technique was found to have certain advantages and was preferred under different circumstances.

The first technique required that sufficient power be applied to maintain the helicopter light on its skids. This placed the engine as high as possible in the power range while maintaining ground contact with the helicopter. This method of power management reduced the time required to accelerate the engine to takeoff power. As takeoff power was applied and the helicopter left the ground, a pitch attitude was assumed such that the helicopter climbed and accelerated simultaneously and reached the desired airspeed while passing through a height of 50 feet above the ground. This technique is referred to as the "simultaneous climb accelerate from light on skids" technique.

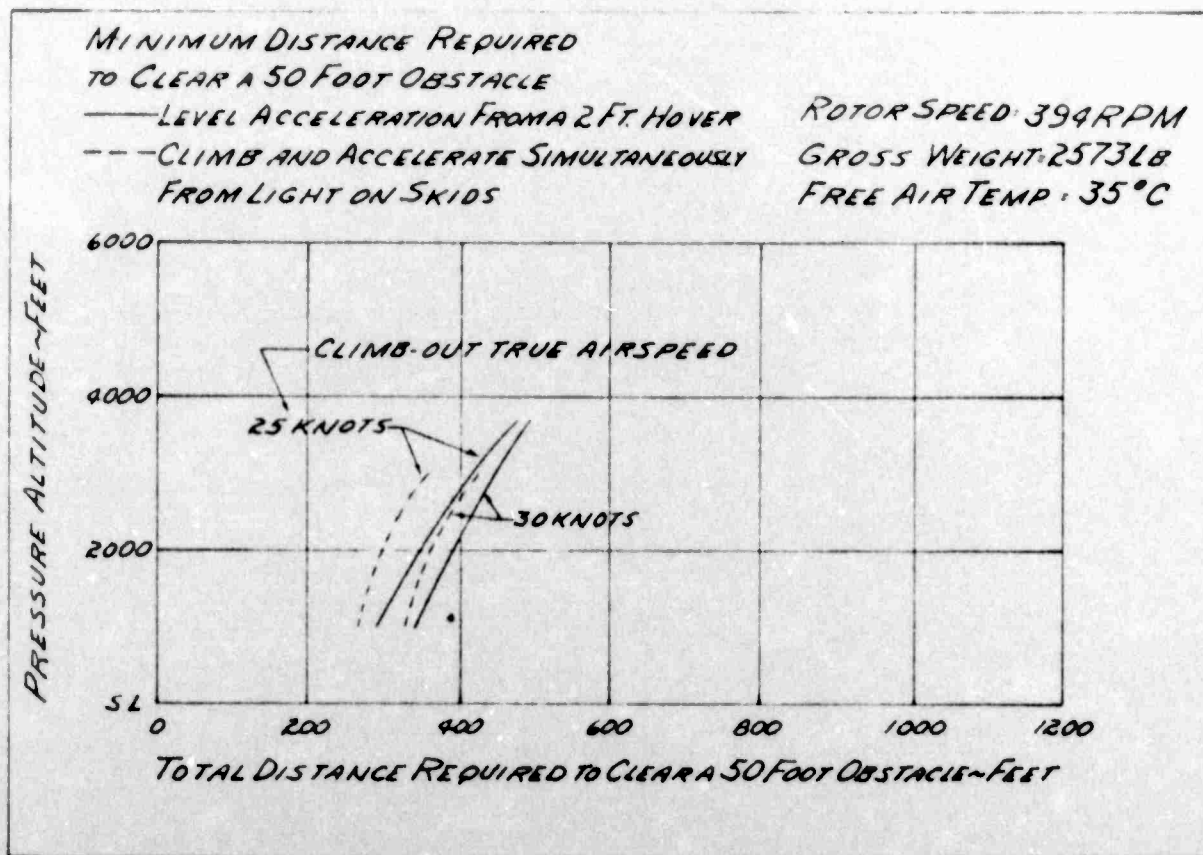
This technique produced shorter takeoff distances at conditions of higher values of excess power (equivalent hovering skid heights of 2-1/2 feet or more). In addition, the technique had several other inherent advantages. A flight path clear of intermediate obstacles was not required. Because of the terrain avoidance features of this technique, the pilot was able to monitor more closely the aircraft instruments, especially with regard to maintaining maximum engine power output and proper rotor speed.

One disadvantage of this technique was that considerable pilot proficiency was required to use it with precision. Attaining the desired airspeed at a 50 foot skid height was entirely dependent upon pilot judgment and skill in assuming the proper initial attitude at lift-off. An additional disadvantage was that the helicopter was within the "avoid" area of the height-velocity curve very early in the takeoff maneuver.

The second takeoff technique was initiated from a 2 foot hovering skid height. As power was applied, the helicopter was accelerated at a constant 2 foot skid height until approximately 3 knots before the desired climb-out airspeed was attained. The helicopter was then rotated to the proper attitude and allowed to climb out at the desired airspeed. When using this technique with a large amount of excess power, pronounced nose-down attitudes were required in order to maintain a constant skid height throughout the acceleration to climb-out airspeed. When there was very little excess power available, the helicopter had a tendency to lose height when passing through translation. If the helicopter were allowed to contact the ground, acceleration decreased and the takeoff distance increased.

Acceleration through translation was accomplished as rapidly as possible. Practice was required to achieve the maximum acceleration attainable for the various amounts of excess power available during the tests.

At large values of excess power available, equivalent to hovering skid heights of 2-1/2 feet or more, the minimum takeoff distances were obtained using the "simultaneous climb and accelerate from light on skids" technique. The following plot indicates that the "level acceleration from a 2 foot hover" technique may produce shorter takeoff distances under some conditions. At high values of excess power, however, the airspeed for minimum takeoff distance was lower for the "simultaneous climb and accelerate from light on skids" technique, and shorter overall distances were required.



Data were not collected for the "climb and accelerate" technique under conditions of low excess power (equivalent hovering skid height of approximately 3 feet or less) and low climb-out-airspeeds

(less than 15 knots). Program termination dates and the available atmospheric conditions for testing precluded gathering these data. There was a minimum value of excess power below which this technique would have no longer been optimum or even practical. Extrapolation of the data in the preceding plot indicated that this point in excess power was at an equivalent hovering skid height of approximately 2-1/2 feet. At lower values of excess power than approximately 2-1/2 feet equivalent hovering skid height, ground effect was required in order to accelerate through translation to the desired climb-out airspeed, and translational lift was necessary before climb-out could be accomplished.

b. Flight Characteristics

The flight characteristics of the OH-4A during takeoff were, in general, satisfactory for achieving optimum takeoff performance. The static longitudinal stability and good controllability about all axes permitted proper attitude control. Control of aircraft attitude along with proper power management was vital to achieving consistent maximum takeoff performance. The importance of attitude control is emphasized because the shortest horizontal distances required to clear a 50 foot obstacle were achieved at airspeeds lower than that at which the standard aircraft's airspeed system became effective, i.e., less than 25 knots true airspeed (KTAS). The takeoff airspeed and the resulting performance were, therefore, entirely dependent upon pilot judgment and proficiency. It should be noted that the takeoff performance presented in this report can only be obtained by a proficient pilot.

Translation to forward flight was comparatively smooth, with no abrupt trim changes. At very low excess power, some settling was evident which occasionally resulted in ground contact. Following translation, the additional lift significantly increased the acceleration of the helicopter to climb-out airspeed. For this reason, it was important to pass through translation as rapidly as possible to obtain maximum performance.

c. Power Management

Power management during maximum performance takeoff in the OH-4A presented difficulties which were detrimental to the overall takeoff performance of the helicopter. An excessive amount of pilot attention was required to monitor engine operation during abrupt power changes.

To achieve maximum takeoff performance, it was important that the takeoff power limit be attained and maintained as rapidly and accurately as possible following initiation of the maneuver. Maintaining the maximum engine power output required that the pilot continuously

make small adjustments to the collective control setting as the aerodynamic loading on the main rotor varied in different flight regimes. This was especially noticed while passing through translation. These corrections in collective pitch caused the rotor speed to vary significantly during a takeoff and from one takeoff to the next under the same start conditions of power and rotor speed.

It was the function of the power turbine (N_2) governor and the droop compensator cam to maintain a nearly constant rotor speed during small variations in power (collective control) settings. This function was not consistently performed, so that it was very difficult to maintain a constant rotor speed throughout the takeoff run. The variation in rotor speed with small power changes was not consistent from one takeoff to the next. The pilot could not effectively anticipate and correct rotor speed changes with the power turbine speed selector "beep" switch. In the course of the takeoff performance testing, an occasional inadvertent rotor overspeed at the takeoff power limit was encountered while attempting to maintain 394 (100 percent) revolutions per minute (rpm). (See Section 3, Appendix II, for a more complete description of fuel control operation.)

During takeoff under conditions of large amounts of available excess power, the gas producer must accelerate through a considerable range. The response characteristics of the T63-A-5 engine installed in the OH-4A were found to be unsatisfactory. The average acceleration time from flight idle (63 percent N_1) to 99.5 percent gas producer speed (N_1) was approximately 5 seconds, which is within acceptable limits. However, the initial acceleration after a rapid throttle application (throttle jam) was slow, with the acceleration rate increasing very rapidly during the last 2 seconds. This acceleration characteristic resulted in a large increase in torque during the last second in the acceleration cycle. This was difficult for the pilot to anticipate and to apply the anti-torque control necessary to counteract the resulting yawing motion of the helicopter.

2.2 HOVER

2.2.1 OBJECTIVE

Hovering tests were conducted to determine hovering performance of the OH-4A helicopter. Data from these tests were used to check compliance with the Military Characteristics and to determine if the hovering contractual guarantees were met.

2.2.2 METHOD

Hovering performance data were obtained using the "free flight" method, with data being recorded at various pre-selected skid heights both in-ground effect (IGE) and out-of-ground effect (OGE). A weighted cord of the desired length was used in conjunction with hand signals from a ground observer to obtain precisely the desired skid height. Data were recorded at stabilized skid heights of 2, 5 and 10 feet (IGE) and 50 feet (OGE) in zero wind conditions. These tests were performed at gross weights up to the maximum overload at a mid center-of-gravity (C.G.) location (Station 102.0) and rotor speed of 374, 382 and 394 rpm. Testing was accomplished at density altitudes from sea level to 10,300 feet.

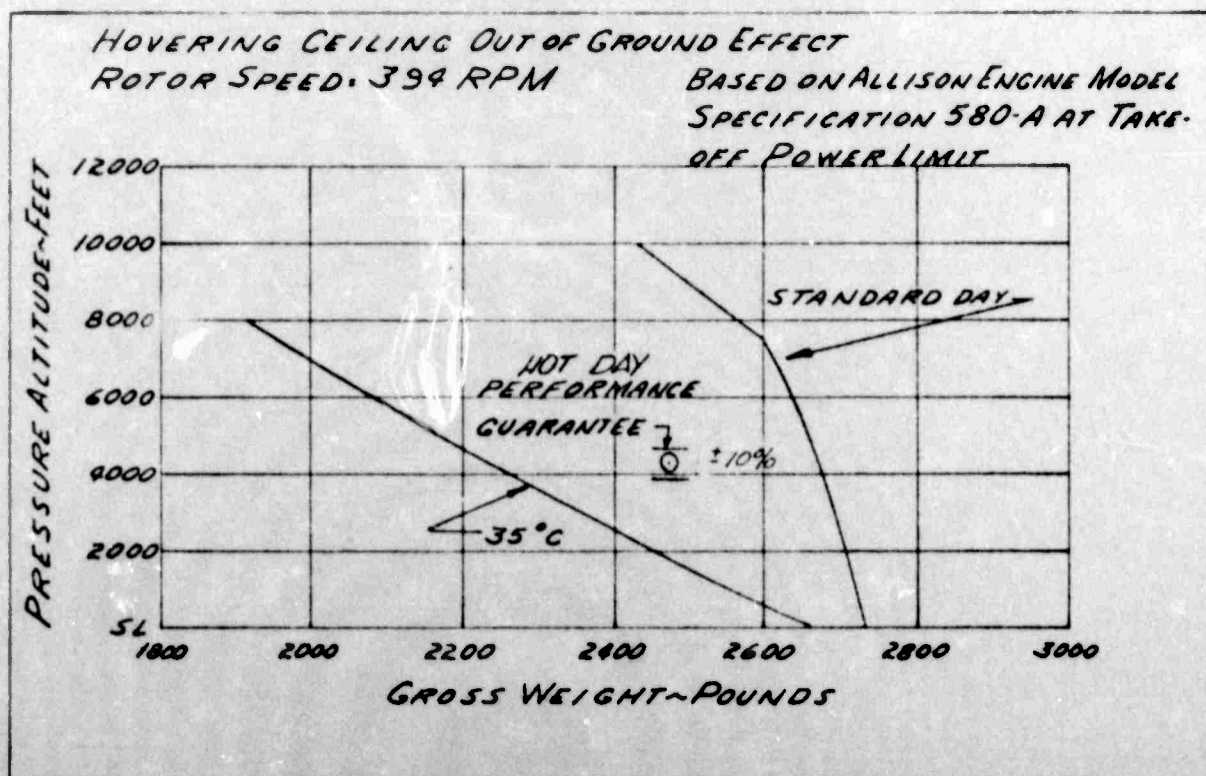
2.2.3 RESULTS

The results of the hovering performance tests are graphically presented in Figures No. 18 through 28, Section 3, Appendix I.

2.2.4 ANALYSIS

a. Guarantee

The OH-4A did not meet the guaranteed hovering performance as shown on the following plot:



The hovering performance guarantee stated that the OGE hovering ceiling at normal gross weight (2573 pounds) with an ambient temperature of 35 degrees centigrade (C) (95 degrees Fahrenheit (F)) would be 4300 feet pressure altitude, \pm 10 percent. Power available was defined by Model Specification 580-A, with installation losses as measured during this evaluation.

The maximum gross weight at which the OH-4A could hover OGE at 4300 feet pressure altitude on a 35 degree C day was 2234 pounds. The actual hovering ceiling under the conditions of the hovering performance guarantee was at a pressure altitude of 825 feet.

The Military Characteristics, Light Observation Aircraft (See Part I, Section 1, paragraph 1.1a) required that the normal gross weight 2573 pounds) OGE hovering ceiling on a 35 degree C day be 6000 feet pressure altitude. The maximum gross weight at which the OH-4A could hover OGE at 6000 feet on a 35 degree C day was 2085 pounds.

The Military Characteristics also required an OGE hovering capability at overload gross weight (2900 pounds) at sea level on a standard day. This requirement was not met using power available (250 SHP) as defined by Model Specification 580-A. The maximum gross weight at which the OH-4A could hover OGE on a standard day at sea level was 2730 pounds. This was 170 pounds (approximately 5.9 percent) less than the requirement.

The summary hovering performance presented in Figure No. 20, Section 3, Appendix I, to check compliance with the performance guarantee, was significantly different from the takeoff power summary hovering performance presented in Figure No. 18, Section 3, Appendix I. The difference was due to the installed power available as defined by the two Model Specifications, 580-A and 580-E. Below critical altitude, the difference was due to the fact that the engine output torque was limited to 218 pound-feet for the 580-A, and to 240 pound-feet for the 580-E. Above critical altitude, the difference was due to the difference in the power available between the two Model Specifications at the same limit turbine outlet temperature (Tt_5). This difference was larger for the higher ambient temperatures.

b. Performance

The hovering performance summaries presented in Figures No. 18 and 19, Section 3, Appendix I, were based on the power available as defined by Model Specification 580-E, with installation losses as measured during this evaluation.

The OGE hovering performance of the OH-4A under standard day sea level conditions was adequate, however, performance deteriorated very rapidly with increasing ambient temperature as well as altitude. The standard day OGE hovering ceiling at the overload gross weight of 2900 pounds was 2200 feet, but on a 35 degree C day, the OH-4A would not hover OGE at sea level at the normal gross weight of 2573 pounds. The maximum gross weight at which the OH-4A could hover OGE at 6000 feet pressure altitude on a 35 degree C day was 1984 pounds.

At normal gross weight (2573 pounds), the 2 foot skid height hovering ceiling was 4150 feet pressure altitude on a 35 degree C day and 12,150 feet on a standard day. At overload gross weight (2900 pounds) the 2 foot skid height hovering ceiling was 1210 feet pressure altitude on a 35 degree C day and 8200 feet on a standard day.

The rapid deterioration of hovering performance with increasing ambient temperature was largely due to the rapid decrease in power available with temperature rise. This power deterioration was considered typical for a helicopter powered by a gas turbine engine when operating above critical altitude.

Comparison of the hovering performance deterioration of the OH-4A and the OH-13H illustrates the difference in power loss with increased temperature for turbine and reciprocating engines. At 8000 feet pressure altitude, above the critical altitude for both helicopters, the maximum gross weight for OGE hovering of the OH-13H was 2532 pounds on a standard day and 2340 pounds on a 35 degree C day (See Section 1, paragraph 1.1p). The maximum gross weight for OGE hovering for the OH-4A was 2495 pounds on a standard day and 1810 pounds on a 35 degree C day at 8000 feet pressure altitude. The OH-13H OGE hovering capability was decreased by 192 pounds or approximately 7.6 percent. The OH-4A OGE hovering capability was decreased by 685 pounds or approximately 27.5 percent.

Rotor speed had a negligible effect upon the OH-4A hovering performance, but rotor speed stability while hovering in gusty winds was poor. The changes in power delivered to the main rotor because of changing aerodynamic loading and control power requirement caused the rotor speed to vary indiscriminately through a range of up to 5 rpm. This variation, however, did not adversely affect skid height or altitude control.

2.3 CLIMBS

2.3.1 OBJECTIVE

Climb tests were conducted to determine the performance during climbing flight and the service ceiling.

2.3.2 METHOD

Continuous climb performance tests were conducted from sea level to service ceiling at two gross weights using maximum continuous power. Takeoff power climbs, for 5 minutes, were also conducted from sea level at the same gross weights. During the climb, power was maintained at limit engine torque until an altitude was reached at which the limiting turbine outlet temperature was obtained (critical altitude). As the climb continued above this altitude, power was adjusted to maintain the limit turbine outlet temperature.

The continuous climbs were conducted at gross weights of 2573 pounds (normal) and 2900 pounds (overload) at a mid C.G. location (Station 102.0) and a rotor speed of 394 rpm.

In addition, sawtooth climbs at normal gross weight and various power settings and maximum continuous climbs at various gross weights and maximum continuous power were conducted to obtain the power and gross weight correction factors.

2.3.3 RESULTS

The results of the climb performance tests are presented graphically in Figures No. 29 through 33, Section 3, Appendix I.

2.3.4 ANALYSIS

The climb performance of the OH-4A was satisfactory. With a start gross weight of 2573 pounds on a standard day at sea level, the service ceiling of the OH-4A was 20,150 feet. At maximum continuous power, the rate of climb at sea level was 1100 feet per minute, increasing to a maximum of 1150 feet per minute at the critical altitude for maximum continuous power (3150 feet). At the same conditions, but using takeoff power, the sea level rate of climb was 1550 feet per minute, increasing to 1575 feet per minute at the critical altitude for takeoff power (2250 feet).

With a climb start gross weight of 2900 pounds at sea level on a standard day, the service ceiling of the OH-4A was 15,600 feet. At maximum continuous power, the sea level rate of climb was 810 feet per minute, increasing to a maximum of 850 feet per minute at the critical altitude for maximum continuous power (3150 feet). At the same conditions, but using takeoff power, the sea level rate of climb was 1250 feet per minute, increasing to a maximum of 1270 feet per minute at the critical altitude for takeoff power (2250 feet).

The airspeed flown to obtain the maximum rate of climb was based upon level flight performance data. Due to time limitations, a complete analysis of level flight data was not possible prior to completing the climb tests, and the optimum climb-speed schedule was not flown in every case. Figures No. 29 and 30, Section 3, Appendix I, show the calibrated airspeed actually flown along with optimum climb speed schedule. At lower altitudes and gross weights, maintaining the airspeed for maximum rate of climb was not critical due to the relatively constant value of shaft horsepower (SHP) required in level flight at true airspeeds near the airspeed for maximum rate of climb. Near the service ceilings, the relationship between SHP required in level flight and true airspeed was more critical, and the actual service ceilings may be slightly higher than those shown in Figures No. 29 and 30, Section 3, Appendix I.

Maintaining the selected airspeed during a climb presented no particular problems once the helicopter was stabilized in the climb. However, at light weight and high power, it was rather difficult to establish a stabilized airspeed at low altitudes. Due to the initial high rate of climb, considerable altitude was gained before the helicopter was completely stabilized. It was found that the best method for stabilizing at the desired airspeed and power setting was to increase power at a rate that allowed the airspeed to remain stabilized.

2.4 LEVEL FLIGHT

2.4.1 OBJECTIVE

Tests were conducted in level flight to determine the range, endurance, speed and power required of the OH-4A helicopter. Data from these tests were used to check compliance with the Military Characteristics and to determine if the contractual guarantees were met. Tests were also conducted to determine the effect on performance of the XM-7 and XM-8 weapons system.

2.4.2 METHOD

Speed power tests were conducted at various conditions of altitude, gross weight and rotor speed in both the unarmed (clean) and armed (XM-7 or XM-8) configurations. Each speed power was flown at a constant value of gross weight divided by density (W/ρ) so that a comparative analysis could be made. This involved increasing altitude as fuel was consumed. During the tests, data were recorded in stabilized level flight at various airspeeds throughout the allowable speed range at approximately 10 knot increments so as to define adequately the particular power required curve. In addition to basic power parameters, fuel flow data were also recorded.

Tests in the clean configuration were conducted at density altitudes of approximately sea level, 5000, 10,000 and 15,000 feet. Gross weights of approximately 2200, 2500 and 2900 pounds were used at a mid center-of-gravity (C.G.) location (Station 102.0). Rotor speeds of 374, 382 and 394 rpm were flown. Two additional tests were conducted at a density altitude of 5000 feet, a gross weight of approximately 2400 pounds, with a rotor speed of 394 rpm. One of these tests was at a forward C.G. location (Station 99.3) and the other at an aft C.G. location (Station 106.3).

Tests in the armed configuration were conducted at density altitudes of approximately 2000 and 6000 feet, a gross weight of approximately 2400 pounds, a mid C.G. location and with a rotor speed of 394 rpm.

2.4.3 RESULTS

The results of the level flight performance tests are presented graphically in Figures No. 34 through 51, Section 3, Appendix I.

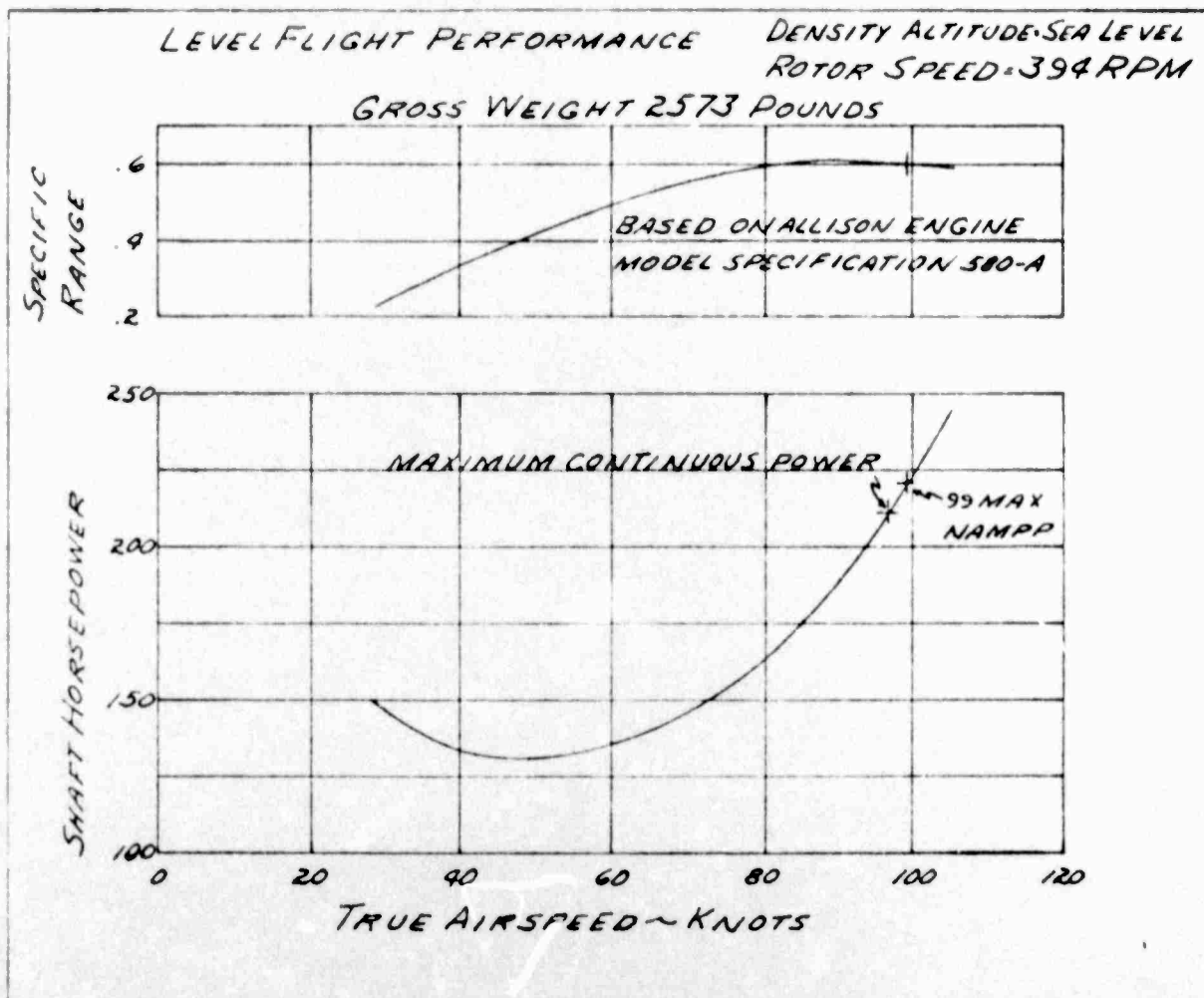
2.4.4 ANALYSIS

a. Guarantee

The OH-4A met the maximum airspeed guarantee. The OH-4A Model Specification maximum airspeed guarantee was 100 knots \pm 10 percent true airspeed for the conditions of sea level on a standard day at normal gross weight (2573 pounds) and maximum continuous power as defined by Model Specification 580-A. The relationship of SHP and specific range to true airspeed at these conditions is presented in the plot on Page 17.

The maximum true airspeed was found to be 96.5 knots. This exceeded the minimum guarantee (100 knots - 10 percent) by 6.5 knots true airspeed (KTAS) (approximately 7.2 percent).

The Military Characteristics, Light Observation Aircraft required a cruise speed of 110 KTAS at the same conditions as above. The OH-4A failed to meet this requirement by a margin of 13.5 knots (approximately 12.3 percent).



The OH-4A failed to meet the endurance mission performance guarantee. The conditions for the endurance mission guarantee were stated in the OH-4A Model Specification as presented in the table on Page 18.

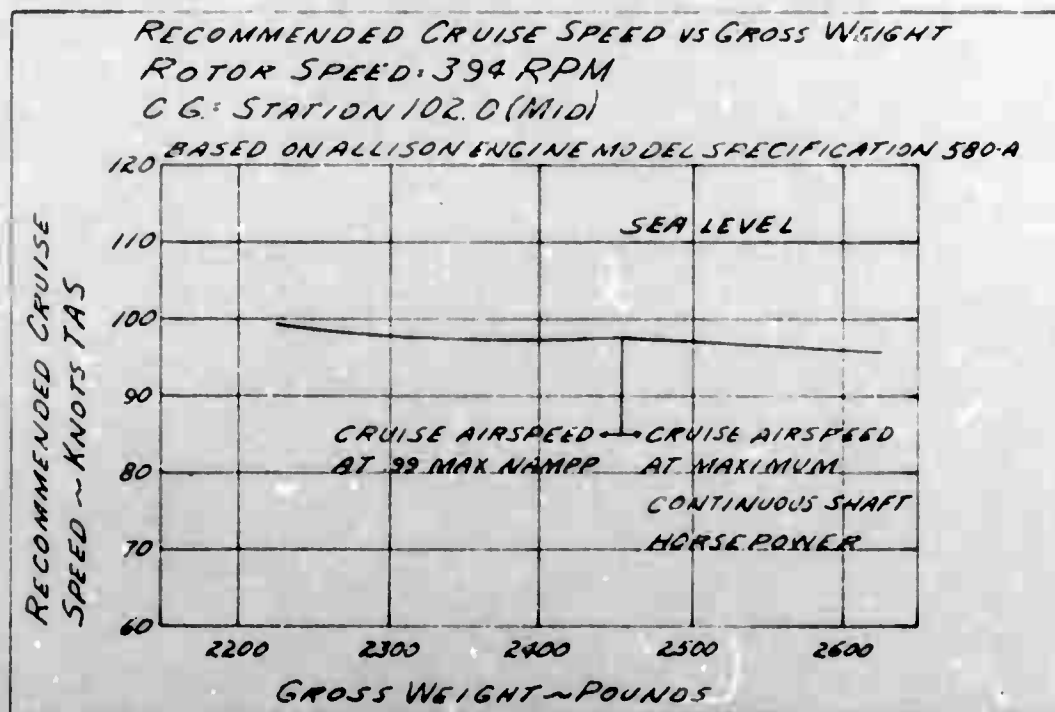
The guaranteed endurance at the conditions stated in Table I, on Page 18, was 3.0 hours \pm 10 percent. The actual endurance of the OH-4A at the above conditions was 2.33 hours. This failed to meet the minimum guaranteed endurance of 2.7 hours (3.0 hours - 10 percent) by a margin of 0.37 hours (approximately 13.6 percent).

TABLE NO. 1

Basic weight (full oil and trapped fuel)	1596
Pilot	200
Observer and cargo	400
Fuel (including 1 lb unusable fuel)	<u>377</u>
Engine start gross weight	2573 lb
Warm up and takeoff: 3 minutes at maximum continuous power at sea level	
Cruise 96 kt at sea level	
No fuel reserve	

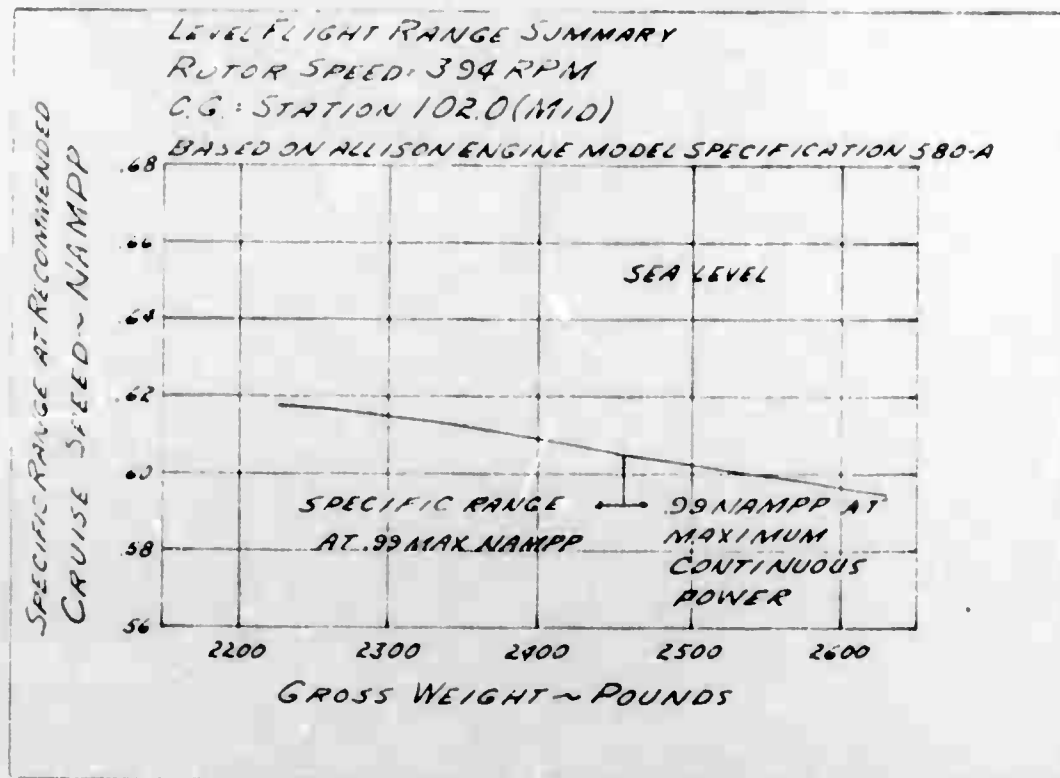
The empty weight of the OH-4A was considerably higher than originally intended. Because of this, the amount of fuel that could be carried had to be reduced to stay within the OH-4A Model Specification normal gross weight of 2573 pounds (2450 pounds + 5 percent). This contributed to a significant decrease in the range and endurance capability of the helicopter.

The airspeed specified for the endurance mission, 96 KTAS, did not represent the airspeed for maximum range performance. The airspeed for maximum range performance, the recommended cruise speed, is shown in the following plot:



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At gross weights greater than 2455 pounds, the cruise speed was limited by maximum continuous power. A specific example of this condition may be seen in the plot on page 17. At gross weights lower than 2455 pounds, the recommended cruise speed was that airspeed at which specific range was 99 percent of its maximum value for a specific gross weight. The values of specific range at the recommended cruise speeds are shown in the following plot:



The Military Characteristics, Light Observation Aircraft (See Part I, Section 1, paragraph 1.1a) listed a requirement for 3 hours endurance with a 400 pound payload. The conditions for this requirement were the same as those for the endurance mission performance guarantee in Table I with the exception of airspeed. The airspeed for minimum power required was used rather than the cruise speed of 96 KTAS. At the airspeed for minimum power required, the endurance of the OH-4A was 3.18 hours, exceeding the requirement by 0.18 hours (approximately 6.0 percent).

A second endurance requirement of the Military Characteristics, Light Observation Aircraft was that at 85 percent cruise power the endurance at the conditions listed in Table I, with the exception of

cruise speed, should be 3 hours. The cruise speed for this requirement was the airspeed obtained at 85 percent of the power required at the recommended cruise speed. The true airspeed at 85 percent cruise power ranged from 86.7 knots at the start of the mission to 89.7 knots with nearly all the fuel used. The endurance at 85 percent cruise power was 2.55 hours. This was 0.45 hours (approximately 15 percent) below the requirement.

b. Performance

The maximum airspeed of the OH-4A was limited either by SHP available or by the "never exceed" airspeed (Vne) as defined by the OH-4A Federal Aviation Agency Type Inspection Authorization (FAA TIA) (See Part I, Section 1, paragraph 1.1g). At density altitudes from sea level to approximately 5000 feet, the limit was usually SHP available. Above that altitude, the limit was usually Vne. The limit encountered depended upon gross weight and ambient temperature as well as density altitude. At a given pressure altitude and gross weight, an increase in ambient temperature would increase the density altitude, and thereby decrease the TIA Vne limit. However, above critical altitude, an increase in temperature would also reduce the power available. The effect of increased temperature upon power available, at a pressure altitude, is greater than the effect upon the TIA Vne. This meant that, at higher ambient temperatures, the pressure altitude at which the TIA Vne limit was reached first was higher than that on a colder day.

The level flight performance tests conducted with the XM-7 armament system installed in the stowed position showed that the equivalent flat plate area of this system was approximately 2.46 square feet. With this increase in drag, the XM-7 installation would reduce the sea level standard day maximum airspeed at normal gross weight (2573) using takeoff power from 111.5 KTAS to 105 KTAS. At maximum continuous power, under the same conditions, the maximum true airspeed would be reduced from 102 knots to 96 knots.

The level flight performance tests conducted with the XM-8 armament system installed in the stowed position showed that the equivalent flat plate area of this system was approximately 1.57 square feet. On a sea level standard day at design gross weight (2573 pounds), the XM-8 armament system reduced the maximum true airspeed at takeoff power from 111.5 knots to 107.5 knots. At maximum continuous power, under the same conditions, the maximum true airspeed would be reduced from 102 knots to 98 knots.

Two level flight tests were conducted at similar conditions of gross weight and density altitude. In one case the helicopter C.G. was at fuselage Station 99.3. In the second case the helicopter C.G. was at fuselage Station 106.3. It was found that there was negligible

difference in the SHP required in level flight at these extremes of C.G. location.

Figure No. 35, Section 3, Appendix I, summarizes the range performance of the OH-4A. The airspeed for maximum range performance for the clean configuration at 394 rpm was a constant 101 KTAS for all gross weights and altitudes tested. The clean configuration includes the F.M. homing antennae, but did not include the XM-7 or XM-8 armament systems. The airspeed for maximum range was not the recommended cruise speed in every case. In some cases the V_{ne} or the airspeed at maximum continuous power limited airspeed to a value lower than that for maximum range performance.

The F.M. homing antennae reduced the range of the OH-4A approximately 1 percent, and reduced the airspeed for maximum range from 103 KTAS to 101 KTAS.

The XM-7 armament system reduced the range approximately 7 percent and reduced the airspeed for maximum range from 101 KTAS to 94 KTAS.

The XM-8 armament system reduced the range approximately 4.5 percent and reduced the airspeed for maximum range from 101 KTAS to 95 KTAS.

Figure No. 68, Section 3, Appendix I, shows the vibration levels for the OH-4A at one gross weight and altitude. Vibration levels were calculated for the fundamental rotor frequency and at the second and fourth harmonics. The one and two-per-rotor revolution vibration levels were less than .07 g's. The four-per-rotor revolution vibration increased at both high and low speeds. Maximum lateral vibration was approximately 0.2 g's at 104.5 KTAS. Maximum vertical vibration was approximately 0.15 g's at the same airspeed. Qualitatively, the overall vibration characteristics of the OH-4A were exceptionally good.

During this evaluation, the OH-4A did not exhibit any tendency to enter blade stall. In the envelope flown during the stability and control portion of this evaluation, there was no tendency toward pitch-up or loss of lateral controllability, the usual indications of incipient blade stall (See Part I). No compressibility effects on the rotor system were detected in the level flight power required tests. The OH-4A, therefore, could effectively use an increase in installed power. Level flight maximum speeds would be increased under conditions where airspeed was limited by SHP available.

The droop characteristics of the OH-4A were unsatisfactory. During power changes the selected rotor speed changed as much as ± 7 rpm.

It is the combined function of the droop compensator cam, the power turbine (N_2) governor and the gas producer speed (N_1) fuel control to schedule fuel flow to the engine. This fuel flow should meet the torque output demanded by the collective pitch control setting while maintaining a nearly constant engine output speed as selected by the pilot.

In practice, the pilot had to continually use the power turbine speed selector "beep" switch when changing collective control setting. When increasing airspeed in level flight from the airspeed for minimum power required to maximum airspeed, the rotor speed would decrease approximately 3 rpm. When transitioning from a climb at take-off power to level flight, the rotor speed would increase approximately 4 rpm. When initiating a partial power descent from level flight at cruise airspeeds, as in approach to a landing, the rotor speed would increase approximately 7 rpm.

The shortcoming in the governing of rotor speed was particularly significant when the helicopter was being controlled from the copilot position. There was no "beep" switch incorporated in the copilot collective control. The copilot had to release one flight control to activate the pilot's "beep" switch, or the pilot had to activate the switch.

Operation of the "beep" switch was satisfactory in the normal operating rpm range. However, a "dead spot" existed at both extremes of the "beep" range (minimum rpm and maximum rpm). This "dead spot" made very small rotor speed adjustments very difficult when operating in these areas.

2.5 AUTOROTATIONAL DESCENTS

2.5.1 OBJECTIVES

Testing was accomplished to determine optimum rotor speeds and airspeeds for autorotational descents. In addition, data were obtained to allow determination of rates and angles of descent during autorotation.

2.5.2 METHOD

Autorotational descents were conducted at various airspeed and rotor speed combinations throughout the allowable ranges. During the descent, time and altitude were recorded so that rate of descent could be determined.

A series of stabilized descents was conducted at a rotor speed of 394 rpm and at various airspeeds between 35 to 65 knots calibrated airspeed (KTAS). From these tests the airspeed for minimum rate of descent was determined. At that airspeed, a second series of descents was conducted at rotor speeds from 374 to 422 rpm.

The autorotational descents were conducted at density altitudes from approximately 5000 to 10,000 feet. Various gross weights from 2180 to 2900 pounds were flown at a mid center-of-gravity (C.G.) location.

2.5.3 RESULTS

The results of the autorotational descent performance tests are presented graphically in Figures No. 52 through 54, Section 3, Appendix 1.

2.5.4 ANALYSIS

The autorotational descent characteristics of the OH-4A were satisfactory and typical for a helicopter of its size and weight. The airspeed for minimum rate of descent was 48.0 KCAS. At this airspeed, the rate of descent varied from 1680 feet per minute at a rotor speed of 422 rpm to 1495 feet per minute at 374 rpm. The minimum angle of descent at a density altitude of 5000 feet and rotor speed of 394 rpm was achieved at 72 knots true airspeed (KTAS). Gross weight and density altitude had little effect upon rate of descent, but a lower rotor speed produced a lower rate of descent at a constant airspeed.

Above a density altitude of approximately 13,000 feet, it was not possible to achieve true autorotational flight with the engine operating. With the twist grip in the flight idle position and the collective pitch control full down, the engine rpm was too high to allow a complete engine-rotor disengagement ("needles split").

Power recoveries from autorotations were impractical and unsafe if initiated close to the ground (during or after the flare) when operating at density altitudes in excess of 3000 feet. This maneuver could be safely accomplished if throttle application were initiated prior to the flare.

2.6 AIRSPPEED CALIBRATION

2.6.1 OBJECTIVE

The objective of these tests was to determine the airspeed position error for both the standard and test airspeed system.

2.6.2 METHOD

The airspeed calibration of the standard and test systems was determined by using the ground speed course method. The aircraft was flown over a measured course at various stabilized airspeeds on reciprocal headings. Airspeeds from approximately 30 to 105 knots calibrated airspeed (KCAS) using approximately 10 knot increments were flown. These tests were conducted at a density altitude of 1260 feet, a gross weight of 2550 pounds, a mid center-of-gravity (C.G.) location (Station 102.0) and a rotor speed of 394 rpm.

2.6.3 RESULTS

The results of the airspeed calibration are presented graphically in Figure No. 67, Section 3, Appendix I.

2.6.4 ANALYSIS

The calibration of the position error of the standard aircraft airspeed system showed good agreement with that presented by the manufacturer. The standard system had no position error at approximately 50 knots indicated airspeed (KIAS). The position error varied linearly from approximately +1-1/2 knots at 28 KIAS to -5 knots at 110 KIAS.

SECTION 3 - APPENDICES

APPENDIX I - TEST DATA

FIGURE No 1
SUMMARY TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-4A USA SIN 62-4202
CLEAN CONFIGURATION

TECHNIQUE: CLIMB AND ACCELERATE
SIMULTANEOUSLY FROM LIGHT
ON SKIDS

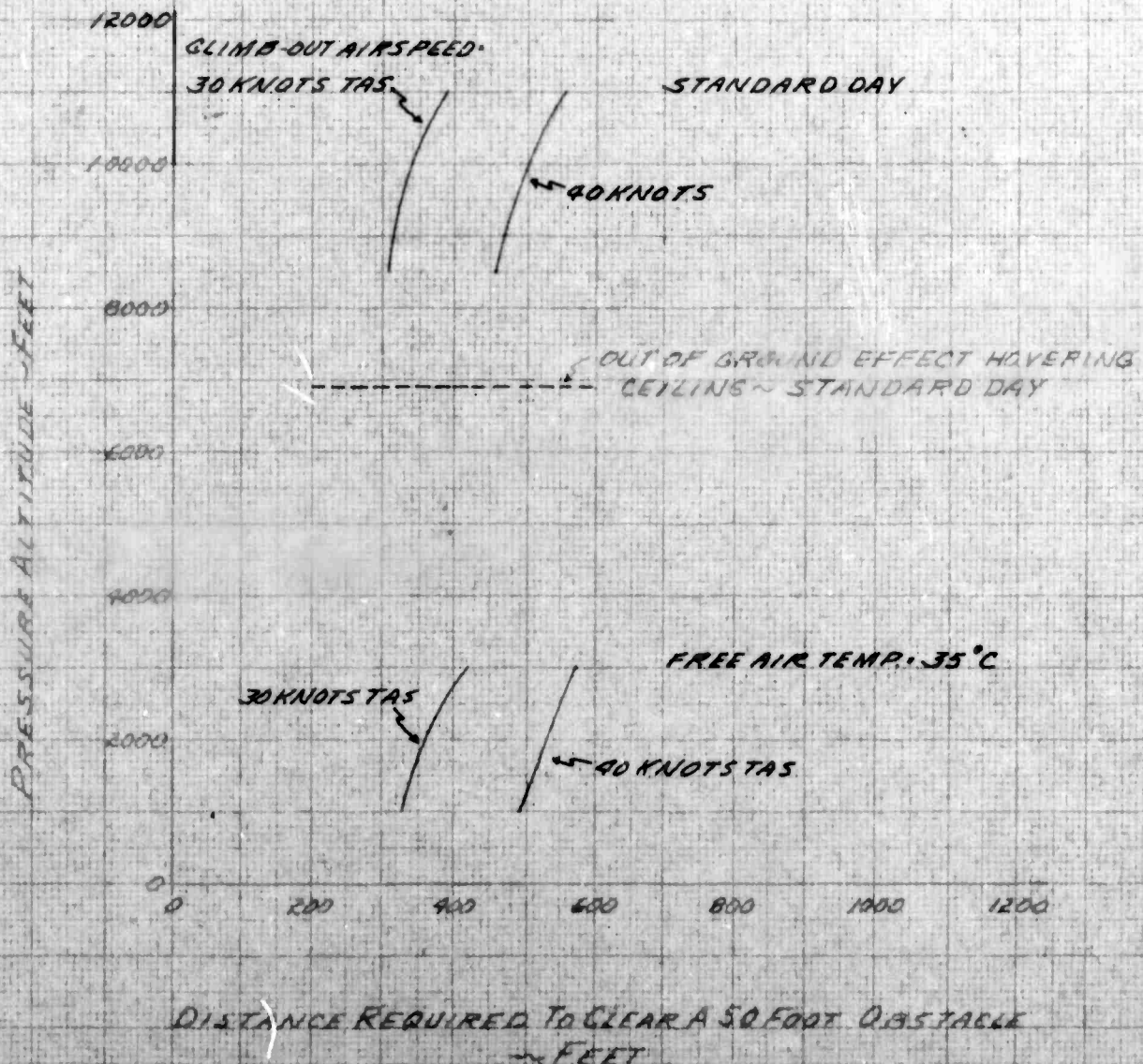
GROSS WEIGHT: 2573 LB

ROTOR SPEED: 394 RPM

--- OUT OF GROUND EFFECT
HOVERING CEILING

NOTE

DERIVED FROM FIGURES
3, 25 AND 55



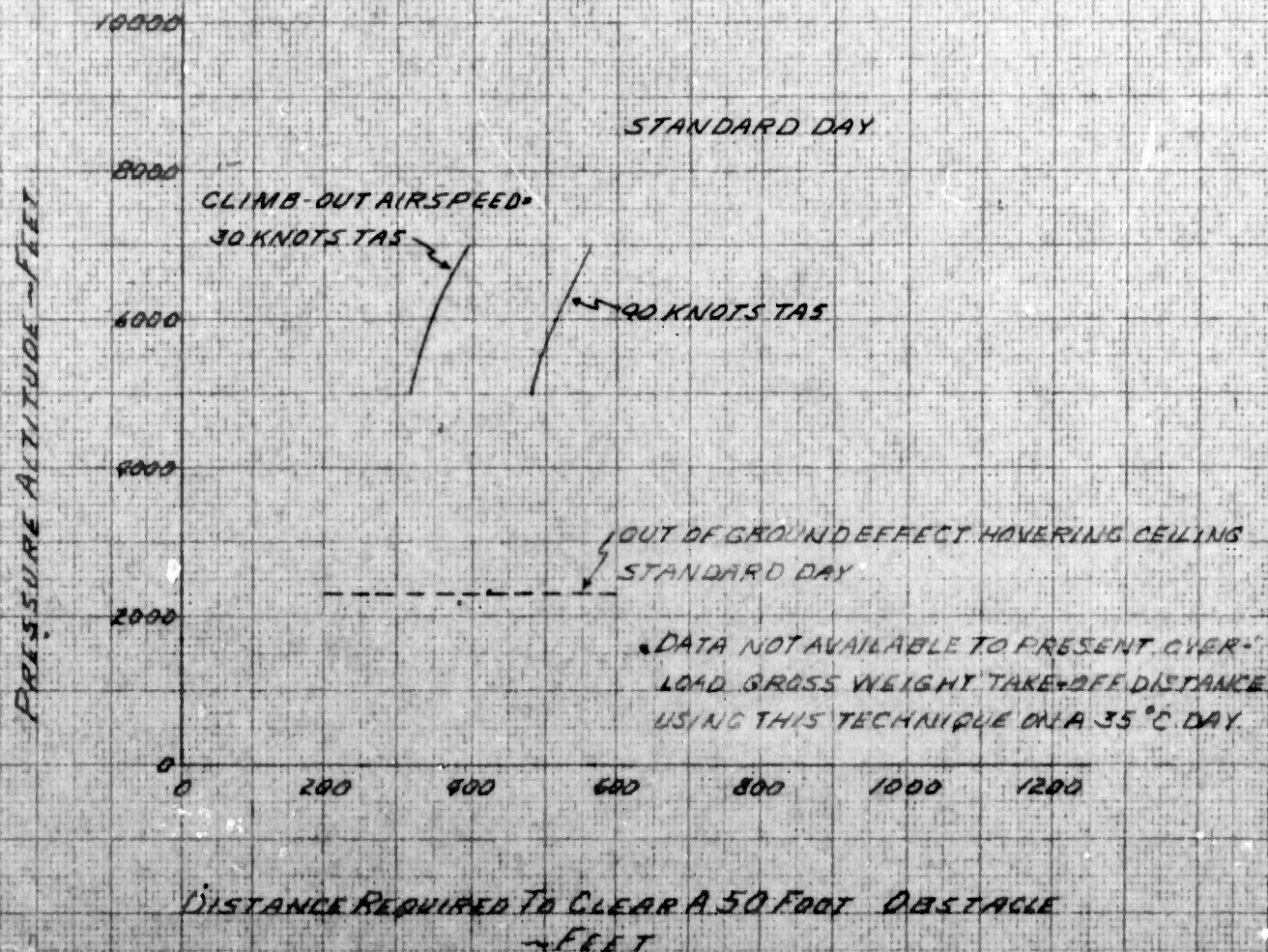
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FORM 10-60
 10-60 TO THE CH. 3201-140

FIGURE No. 2
 SUMMARY TAKE-OFF DISTANCE REQUIRED
 TO CLEAR A 50' OBSTACLE
 OH-9A USA SING2-4202
 CLEAN CONFIGURATION

TECHNIQUE: CLIMB AND ACCELERATE
 SIMULTANEOUSLY FROM LIGHT
 ON SKIDS
 GROSS WEIGHT: 2900 LB
 ROTOR SPEED: 394 RPM
 --- OUT OF GROUND EFFECT
 HOVERING CEILING

NOTE:
 DERIVED FROM FIGURES
 3, 25 AND 55



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FIGURE NO. 3
 VARIATION OF TAKE-OFF DISTANCE
 WITH CLIMB-OUT AIR SPEED AND ΔC_p
 OH-4A USA SINGR-4202
 CLEAN CONFIGURATION ROTOR SPEED: 394 RPM

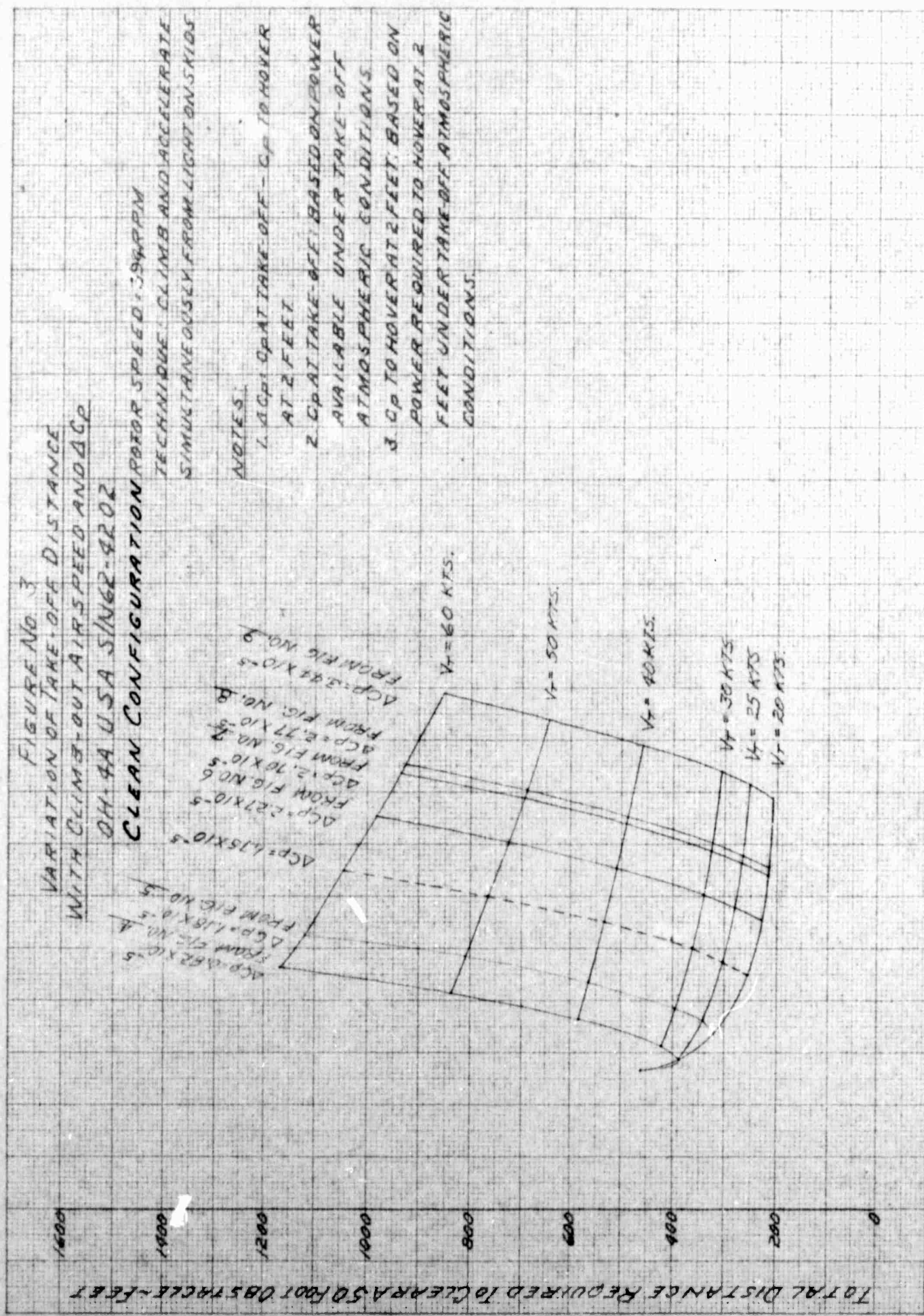


FIGURE NO. 4
 TAKE-OFF DISTANCE REQUIRED
 TO CLEAR A 50' OBSTACLE
 OH-4A USA SING 2-4202
 CLEAN CONFIGURATION
 ROTOR RPM: 394

GROSS WEIGHT: 2292 LB.

FREE AIR TEMP: 7.3°C

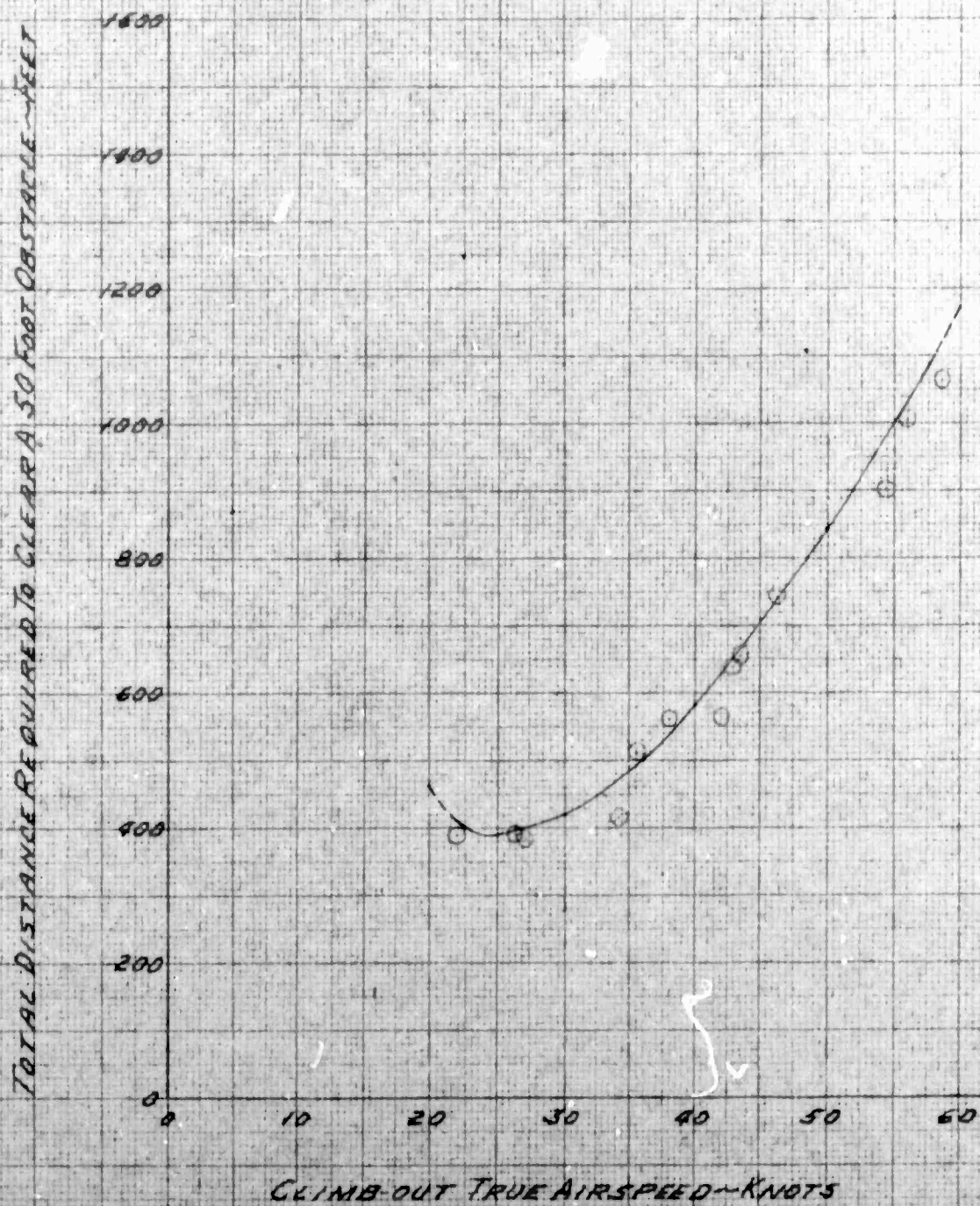
PRESSURE ALTITUDE: 9350 FT.

WIND VELOCITY: 5.3 KNOTS

$\Delta C_p = 0.82 \times 10^{-5}$

TECHNIQUE: CLIMB AND ACCELERATE

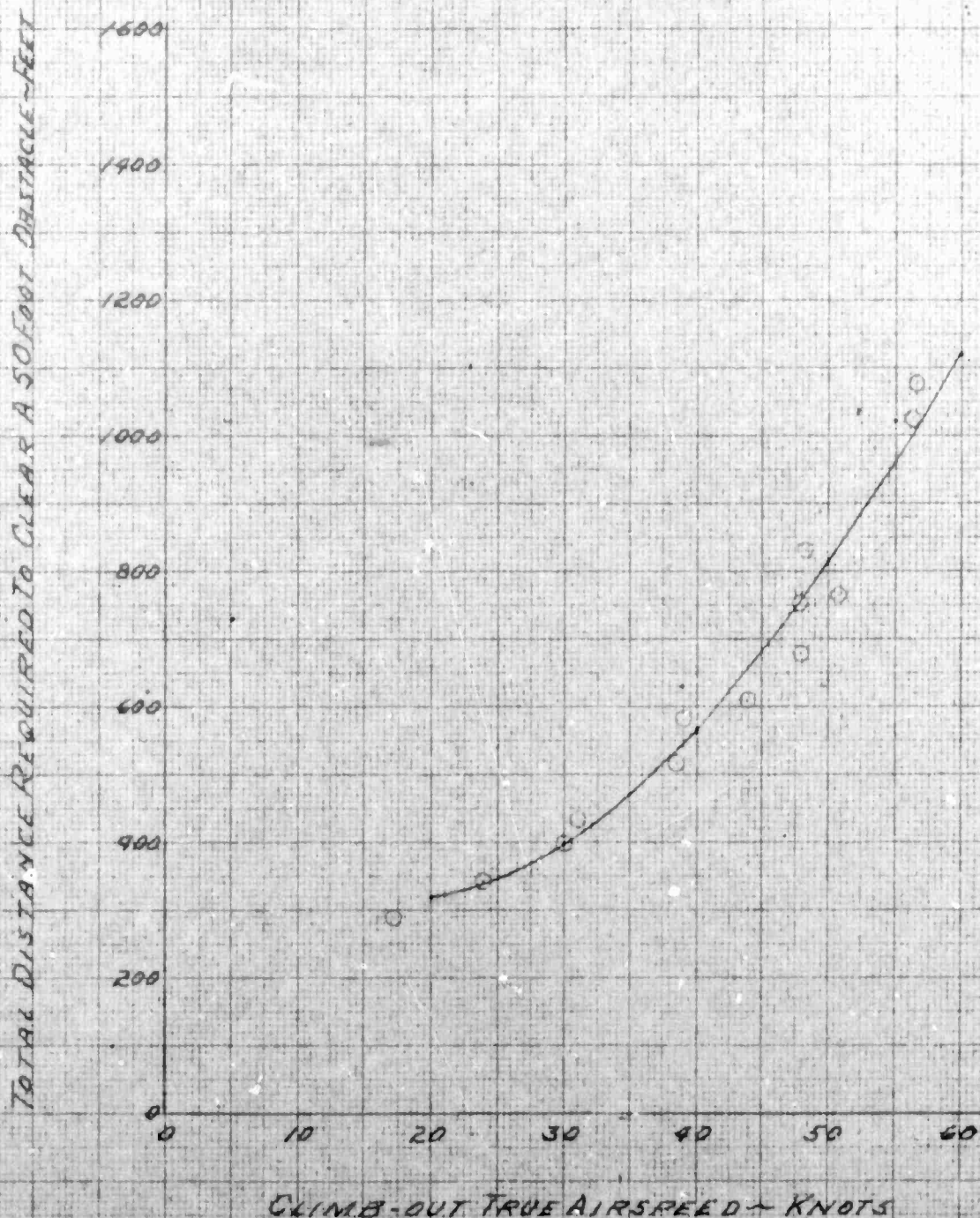
SIMULTANEOUSLY FRONT LIGHT ON SKIDS



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FIGURE NO. 5
 TAKE-OFF DISTANCE REQUIRED
 TO CLEAR A 50' OBSTACLE
 OH-4A USA S/N 62-4202
 CLEAN CONFIGURATION
 ROTOR RPM = 395
 GROSS WEIGHT = 2930 LB.
 FREE AIR TEMP = 10.1 °C
 PRESSURE ALTITUDE = 3750 FT.
 WIND VELOCITY = 4.3 KNOTS
 $\Delta C_p = 1.18 \times 10^{-5}$

TECHNIQUE CLIMB AND ACCELERATE
 SIMULTANEOUSLY FROM LIGHT ON SKIDS



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FIGURE NO. 6
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE

OH-4A USA S/N 62-4202

CLEAN CONFIGURATION

ROTOR RPM = 394

GROSS WEIGHT = 2576 LB

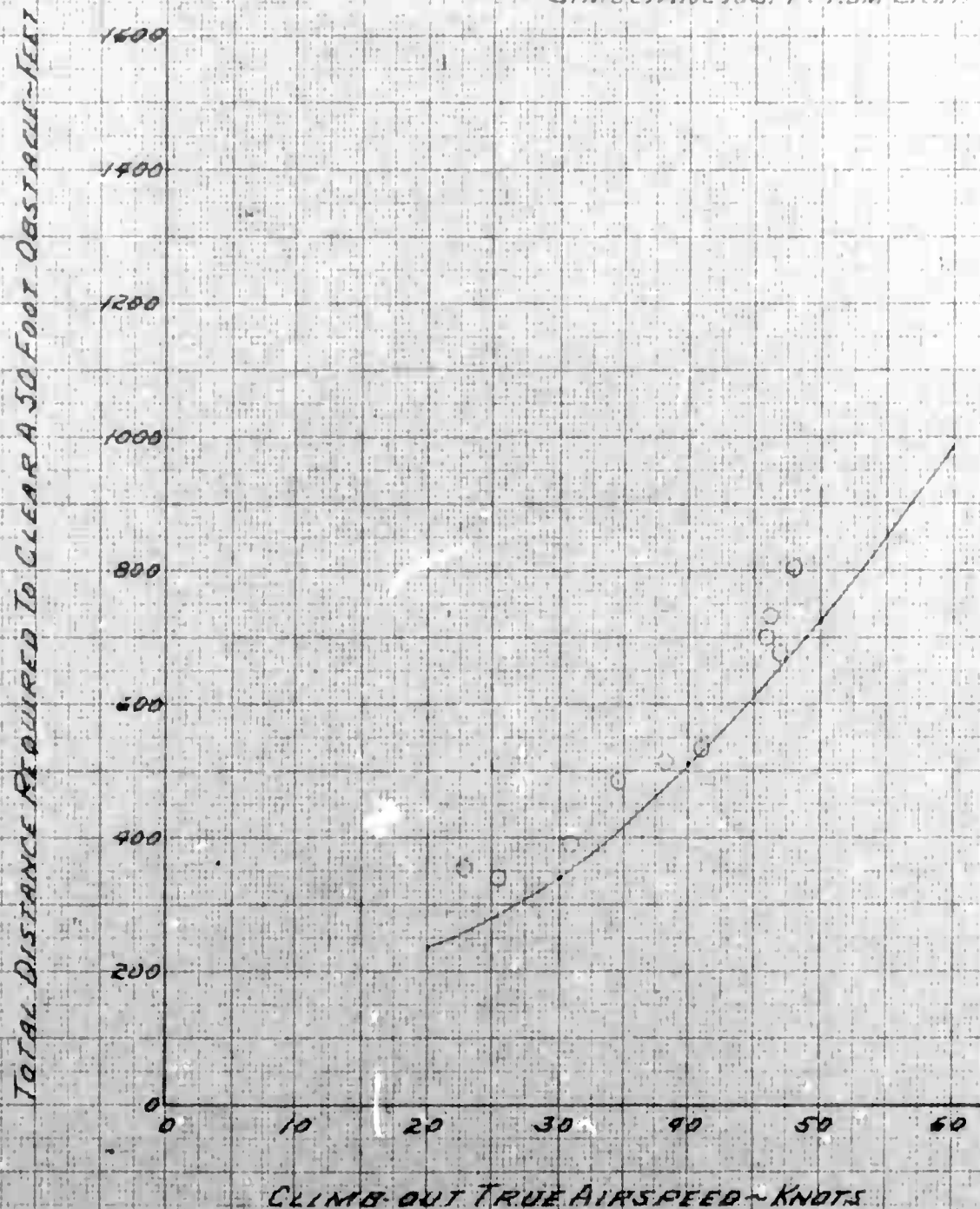
FREE AIR TEMP = 26.8°C

PRESSURE ALTITUDE = 3970 FT.

WIND VELOCITY = 4.3 KNOTS

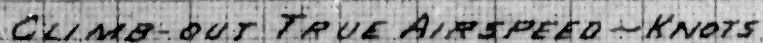
$\Delta C_p = 2.27 \times 10^{-5}$

TECHNIQUE: CLIMB AND ACCELERATE
SIMULTANEOUSLY FROM LIFT OFF SPEEDS



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SIMULTANEOUSLY FROM LIGHT ON SKIOS



FOR OFFICIAL USE ONLY

SIMULTANEOUSLY FROM LIGHT ON SKIDS

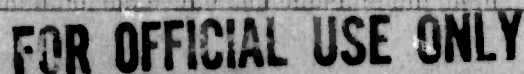


FIGURE No. 9
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE

OH-4A USA S/N 62-4202

CLEAN CONFIGURATION

ROTOR RPM = 394

GROSS WEIGHT = 2276 LB.

FREE AIR TEMP = 22.7°C

PRESSURE ALTITUDE = 3940 FT.

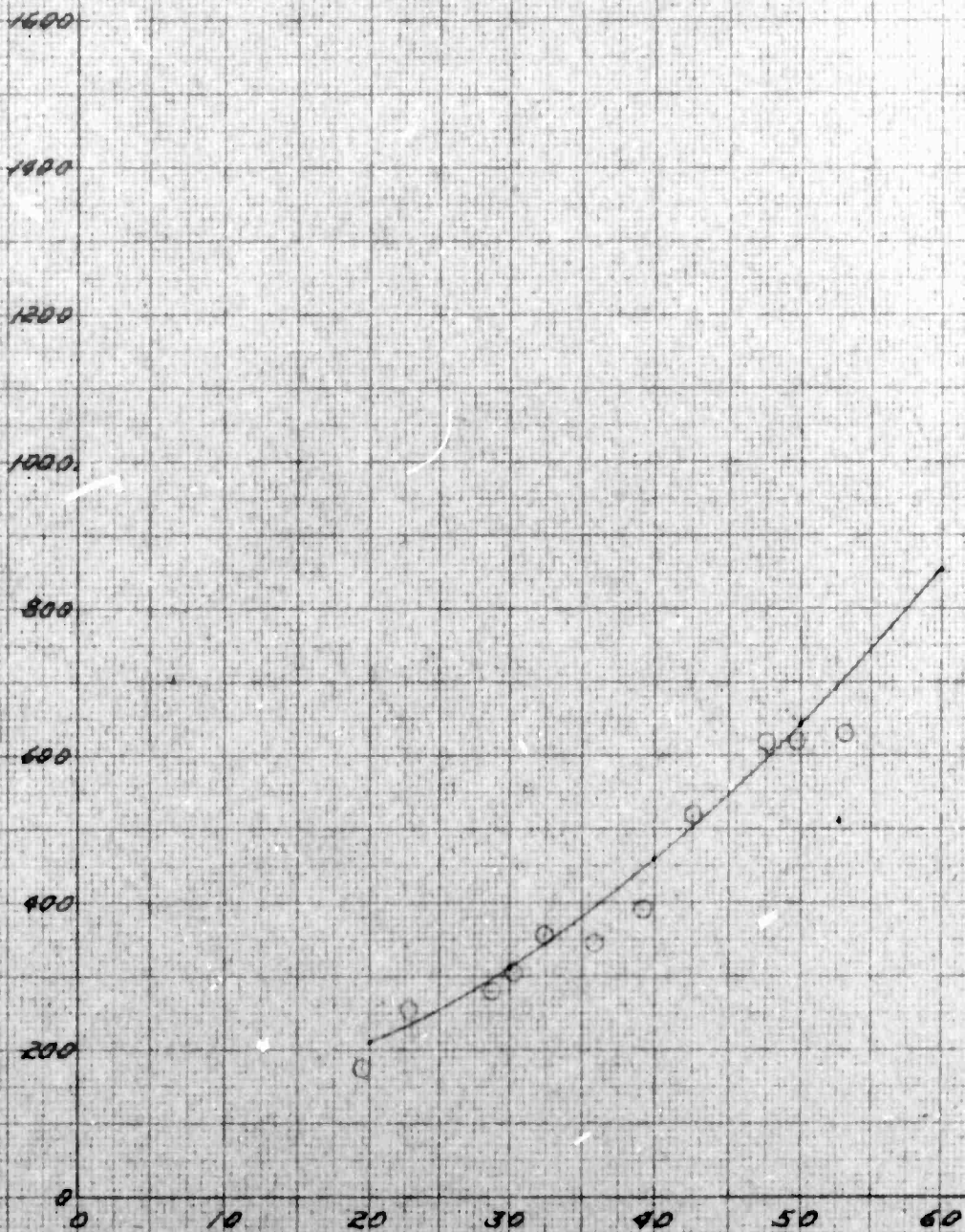
WIND VELOCITY = 6.3 KNOTS

$AC_p = 3.44 \times 10^{-5}$

TECHNIQUE: CLIMB AND ACCELERATE

SIMULTANEOUSLY FROM LIFT OFF ON SKIDS

TOTAL DISTANCE REQUIRED TO CLEAR A 50 FOOT OBSTACLE - FEET



CLIMB-OUT TRUE AIRSPEED - KNOTS

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K-E

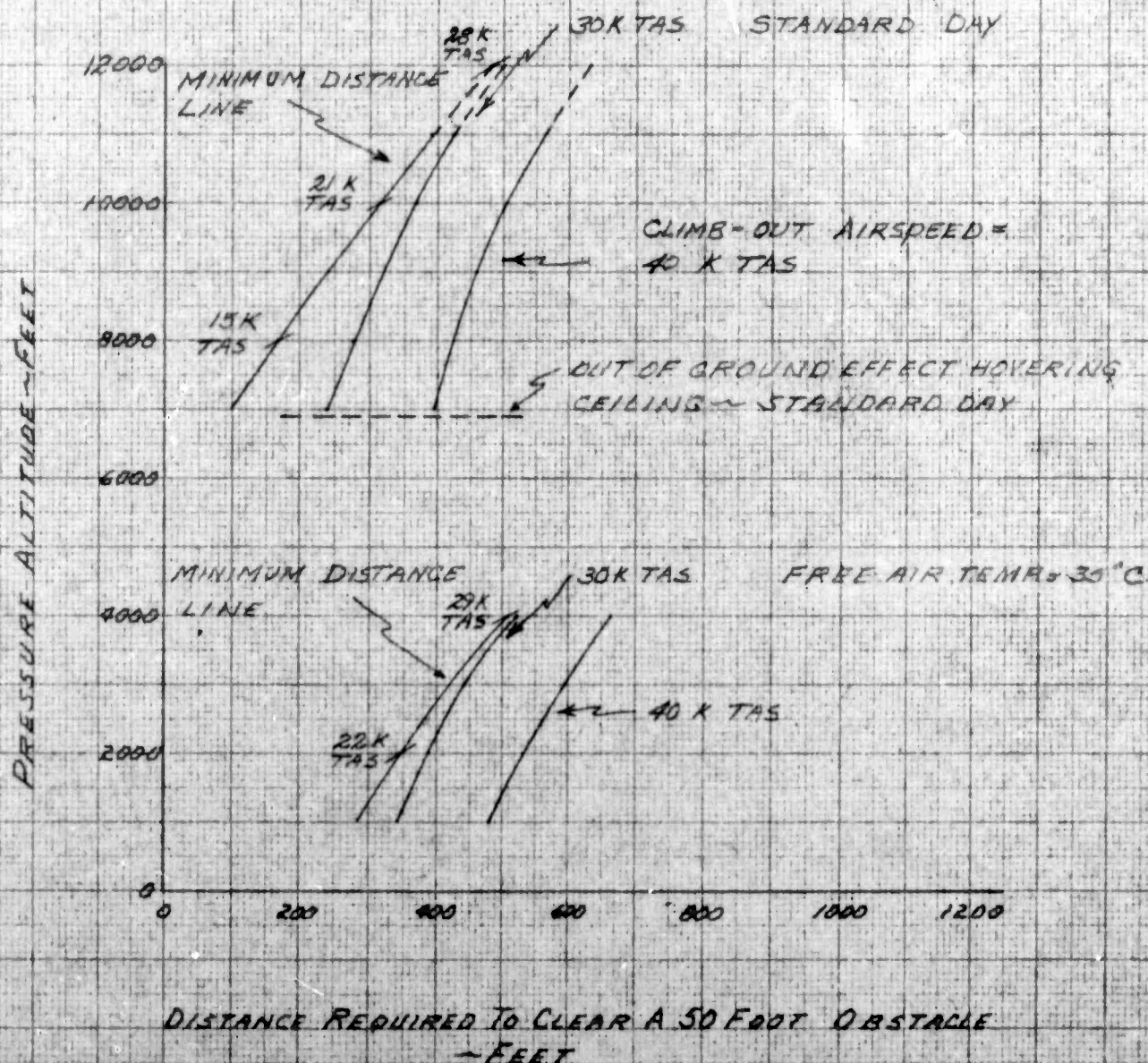
REPLACES 3-1288-10
10 X 10 TO THE CM
3291-140

FIGURE NO. 10
 SUMMARY TAKE-OFF DISTANCE REQUIRED
 TO CLEAR A 50' OBSTACLE
 OH-4A USA SIN 62-9202
 CLEAN CONFIGURATION

TECHNIQUE: LEVEL ACCELERATION
 FROM A 2 FOOT HOVER
 GROSS WEIGHT - 2573 LB
 ROTOR SPEED - 394 RPM
 --- OUT OF GROUND EFFECT
 HOVERING

NOTE

DERIVED FROM FIGURES
 12, 25 AND 53



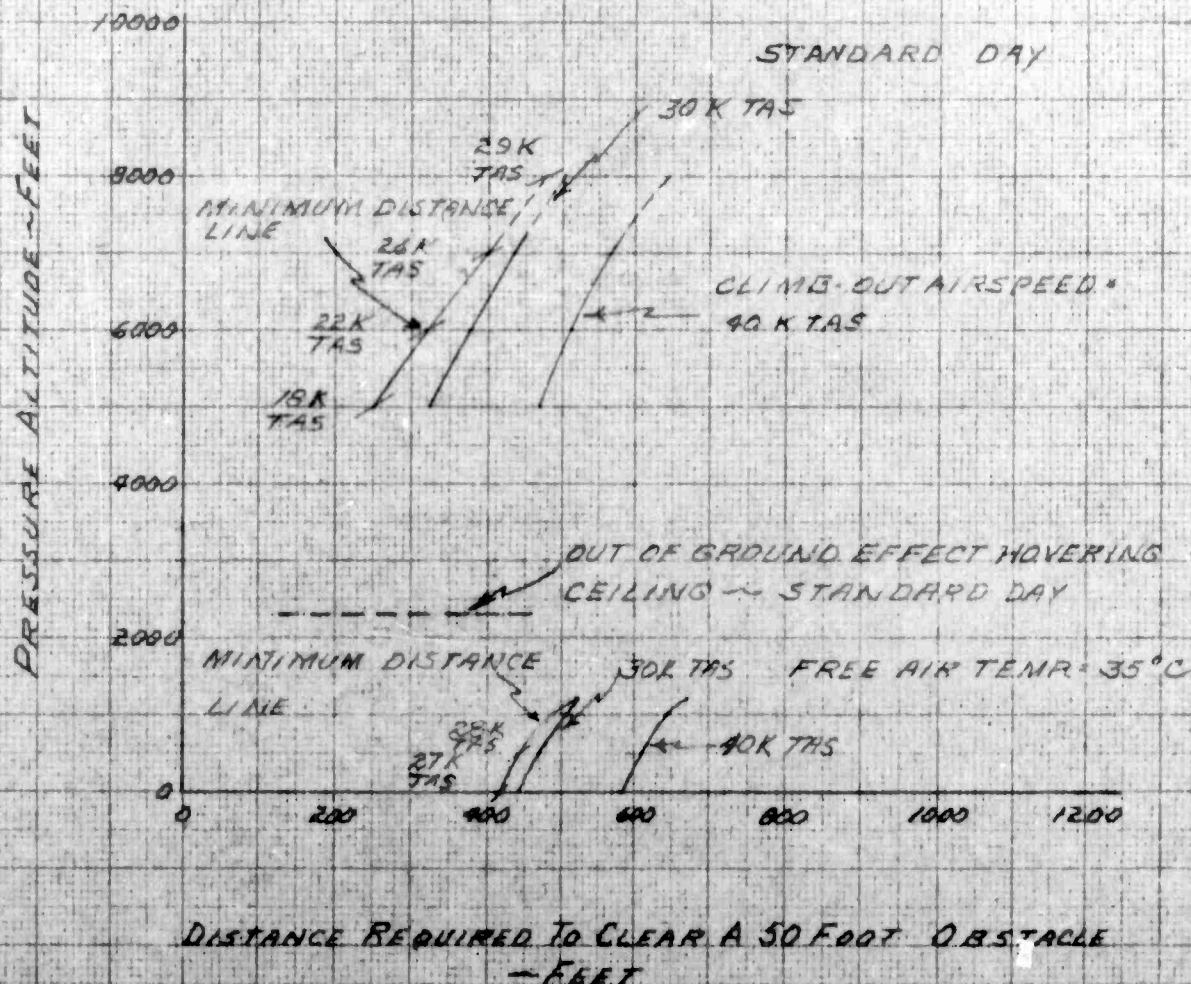
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FIGURE NO. 11
 SUMMARY TAKE-OFF DISTANCE REQUIRED
 TO CLEAR A 50' OBSTACLE
 OH-4A USA SIN 62-9202
 CLEAN CONFIGURATION

TECHNIQUE: LEVEL ACCELERATION
 FROM A 2 FOOT HOVER
 GROSS WEIGHT - 2900 LB.
 ROTOR SPEED - 394 RPM
 --- OUT OF GROUND EFFECT
 HOVERING CEILING

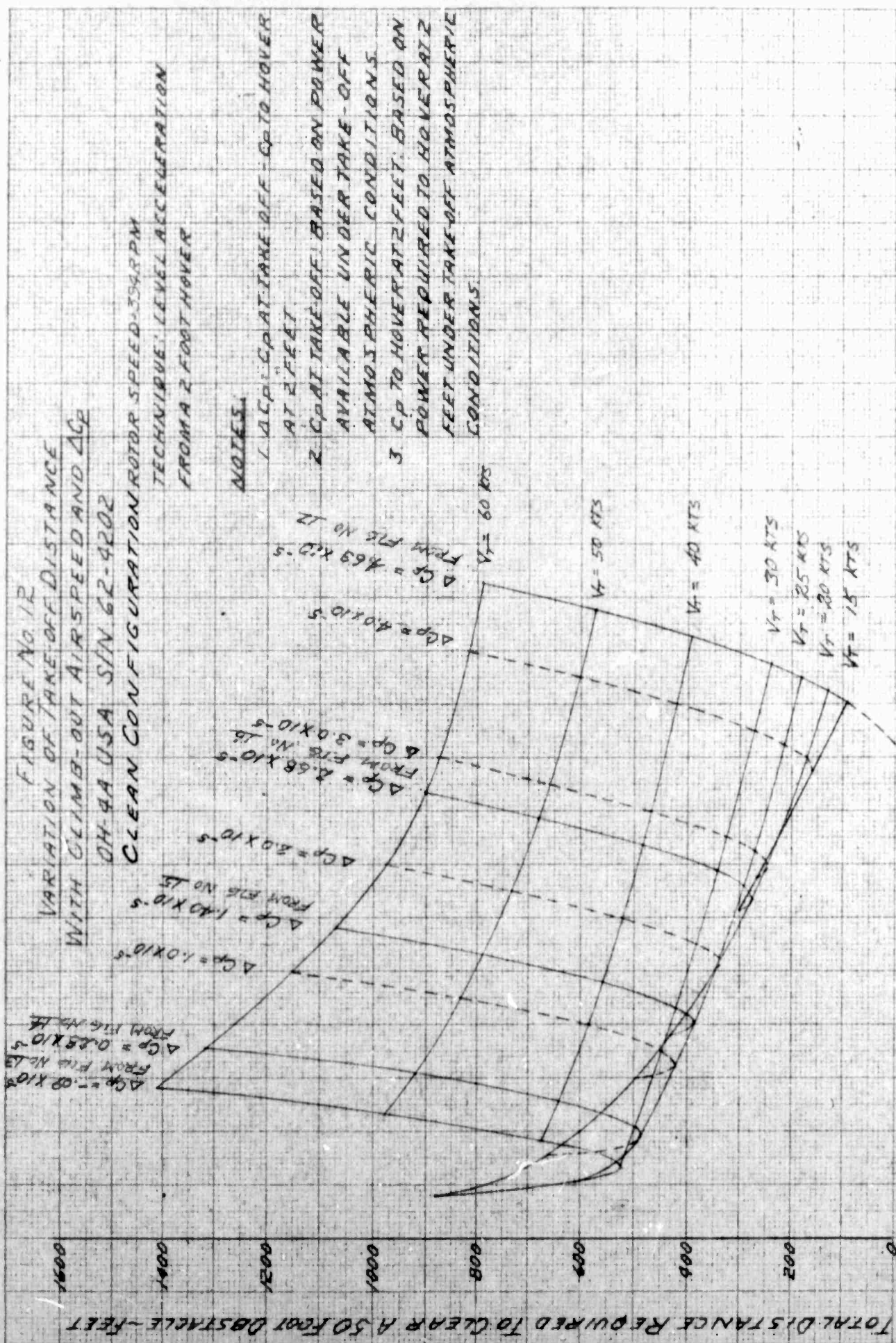
NOTE:

DERIVED FROM FIGURES
 18, 25 AND 55



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FIGURE NO 12
 VARIATION OF TAKE-OFF DISTANCE
 WITH CLIMB-OUT AIRSPEED AND ΔC_p
 OH-4A USA SIN 62-4202
 CLEAN CONFIGURATION ROTOR SPEED 354 RPM



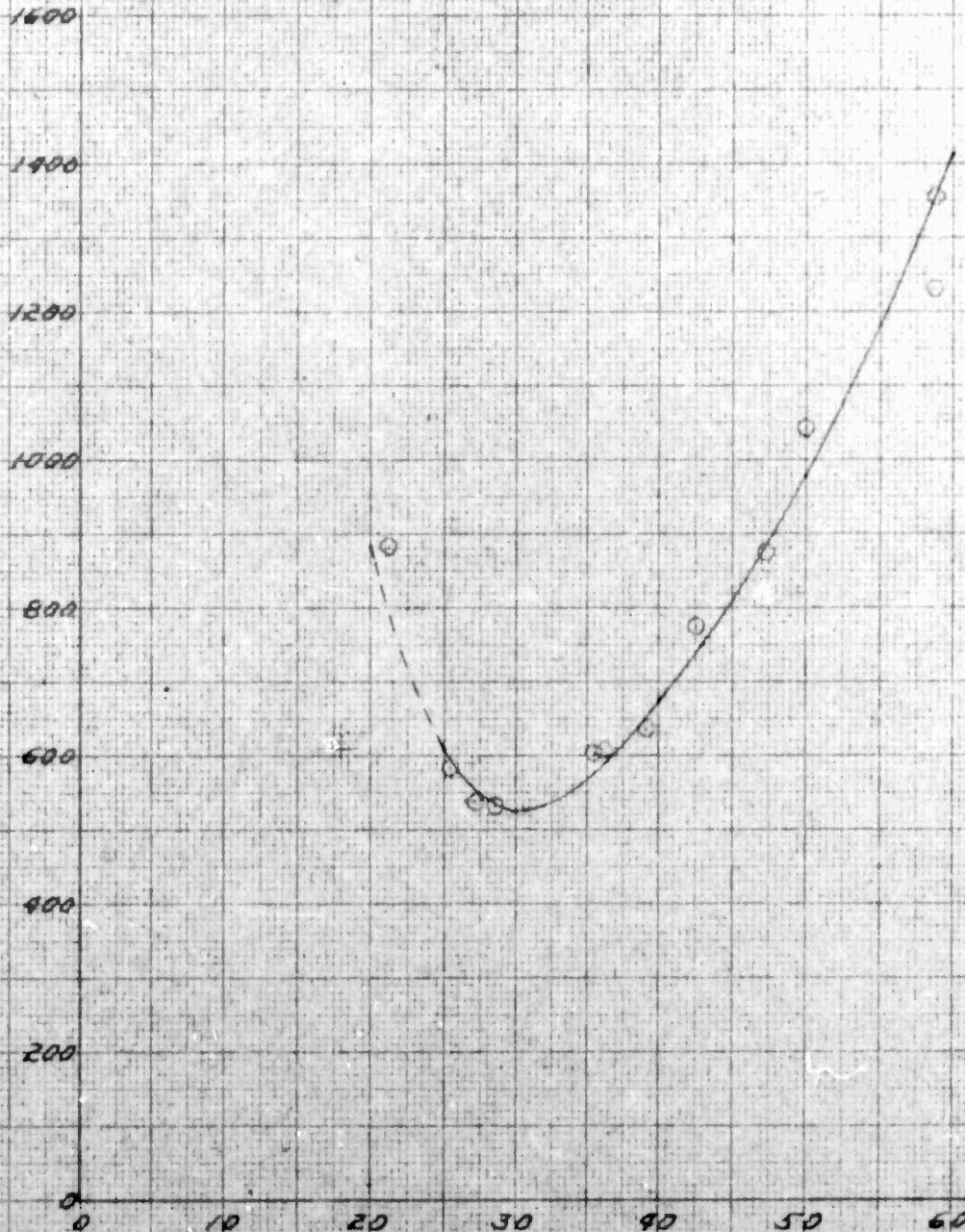
TECHNIQUE: LEVEL ACCELERATION
 FROM A 2 FOOT HOVER

NOTES:

1. ΔC_p : C_p AT TAKE-OFF - C_p TO HOVER AT 2 FEET
2. C_p AT TAKE-OFF: BASED ON POWER AVAILABLE UNDER TAKE-OFF ATMOSPHERIC CONDITIONS
3. C_p TO HOVER AT 2 FEET: BASED ON POWER REQUIRED TO HOVER AT 2 FEET UNDER TAKE-OFF ATMOSPHERIC CONDITIONS

FIGURE NO 13
 TAKE-OFF DISTANCE REQUIRED
 TO CLEAR A 50' OBSTACLE
 OH-4A USA SIN 62-4202
 CLEAN CONFIGURATION
 ROTOR RPM = 393
 GROSS WEIGHT = 2317 LB
 FREE AIR TEMP = 11.6°C
 PRESSURE ALTITUDE = 9290 FT
 WIND VELOCITY = 5.3 KNOTS
 $\Delta C_p = -.09 \times 10^{-5}$
 TECHNIQUE: LEVEL ACCELERATION FROM A 2' HOVER

TOTAL DISTANCE REQUIRED TO CLEAR A 50' OBSTACLE - FEET



CLIMB-OUT TRUE AIRSPEED - KNOTS

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TECHNIQUE: LEVEL ACCELERATION FROM A 2' HOVER



FIGURE No 15
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE

OH-4A USA SIN 62-4202

CLEAN CONFIGURATION

ROTOR RPM = 394

GROSS WEIGHT = 2536 LB.

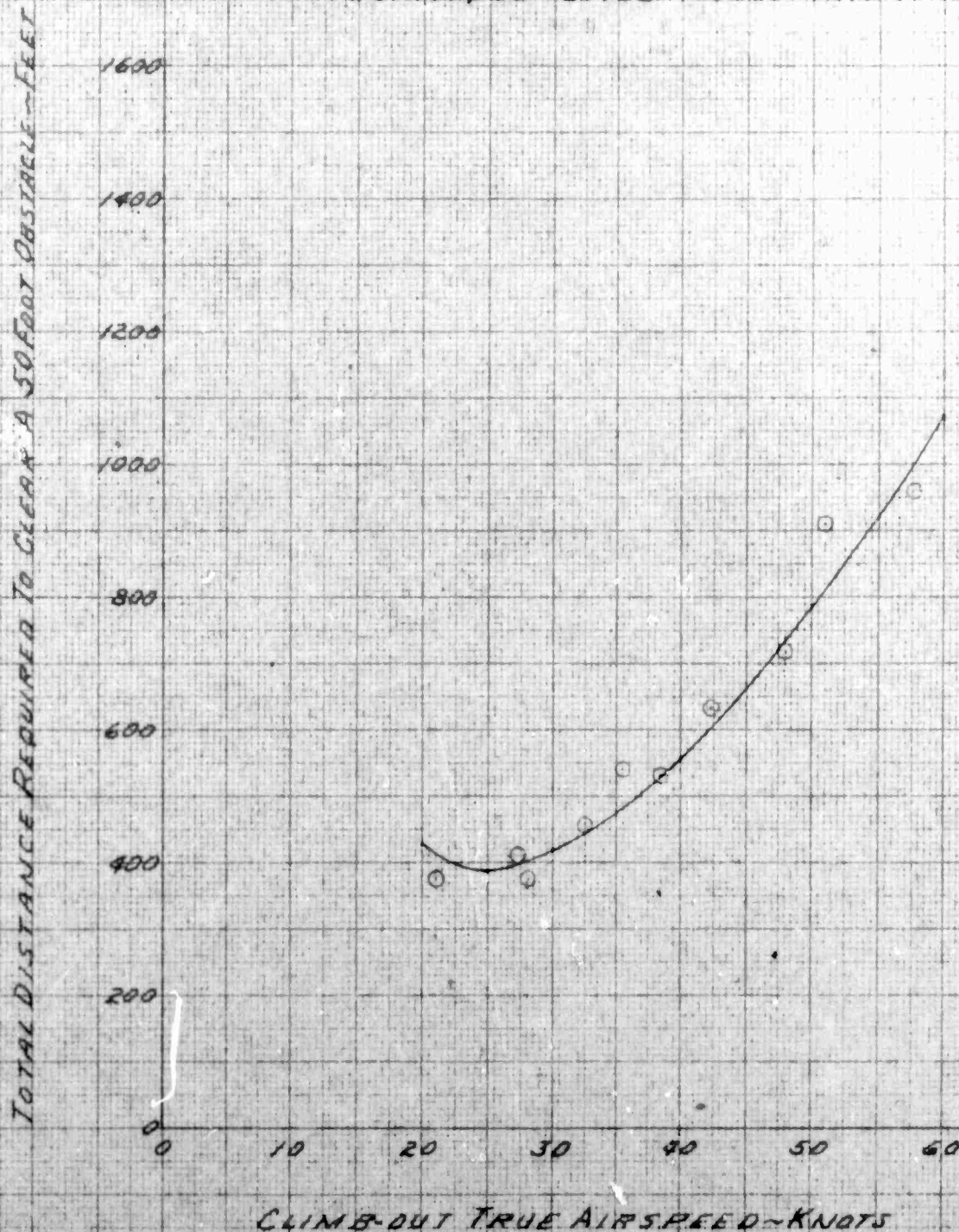
FREE AIR TEMP = 26°C

PRESSURE ALTITUDE = 3930 FT.

WIND VELOCITY = 5.3 KNOTS

$\Delta C_p = 1.40 \times 10^{-5}$

TECHNIQUE: LEVEL ACCELERATION FROM 12' HOVER



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FIGURE NO. 16
TAKE-OFF DISTANCE REQUIRED
TO CLEAR 50' OBSTACLE

OH-4A USA S/N 62-4202

CLEAN CONFIGURATION

ROTOR RPM = 394

GROSS WEIGHT = 2415 LB.

FREE AIR TEMP = 22.9°C

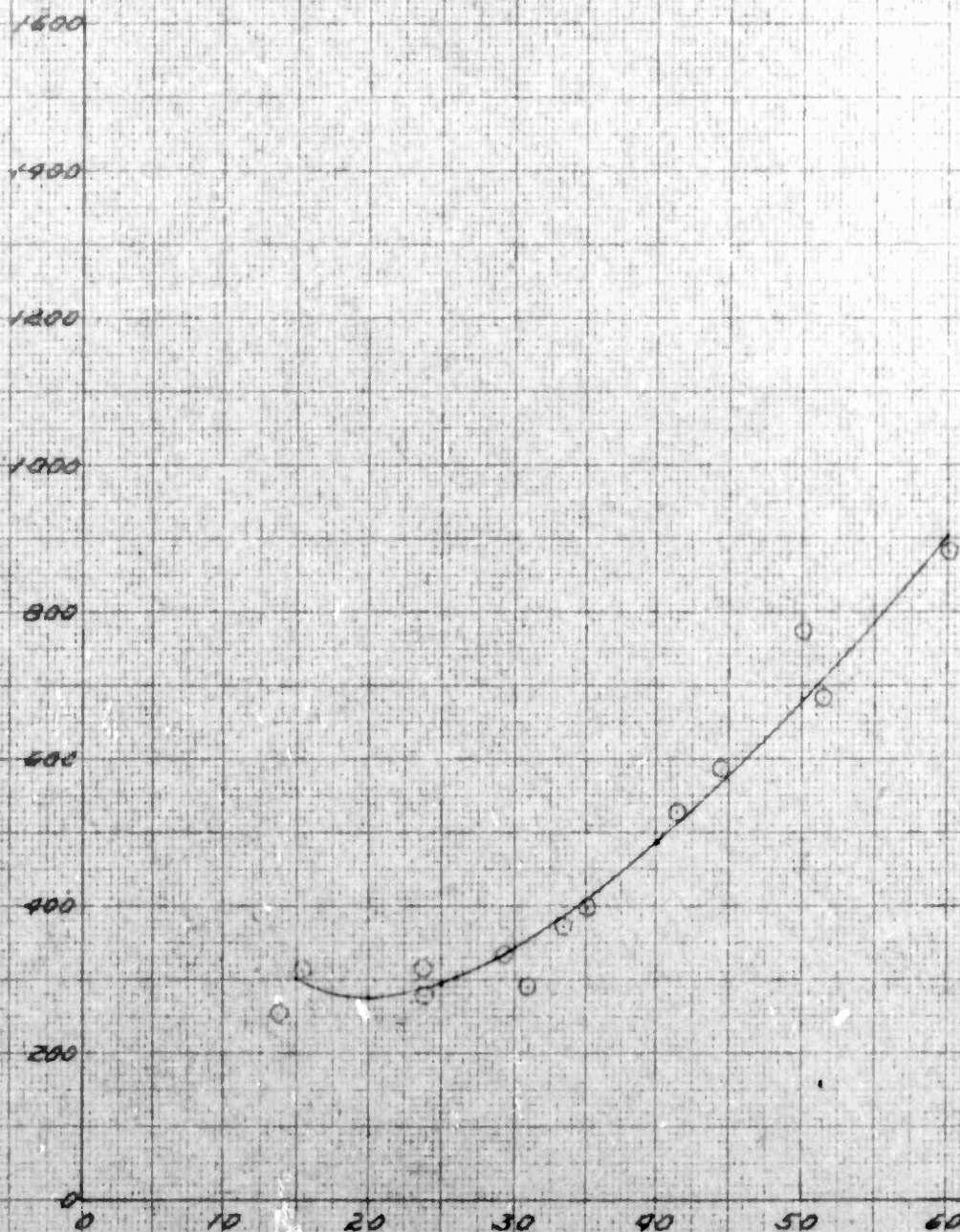
PRESSURE ALTITUDE = 4120 FT.

WIND VELOCITY = 5.3 KNOTS

$\Delta C_p = 2.68 \times 10^{-5}$

TECHNIQUE: LEVEL ACCELERATION FROM A 2' HOVER

TOTAL DISTANCE REQUIRED TO CLEAR 50 FOOT OBSTACLE - FEET

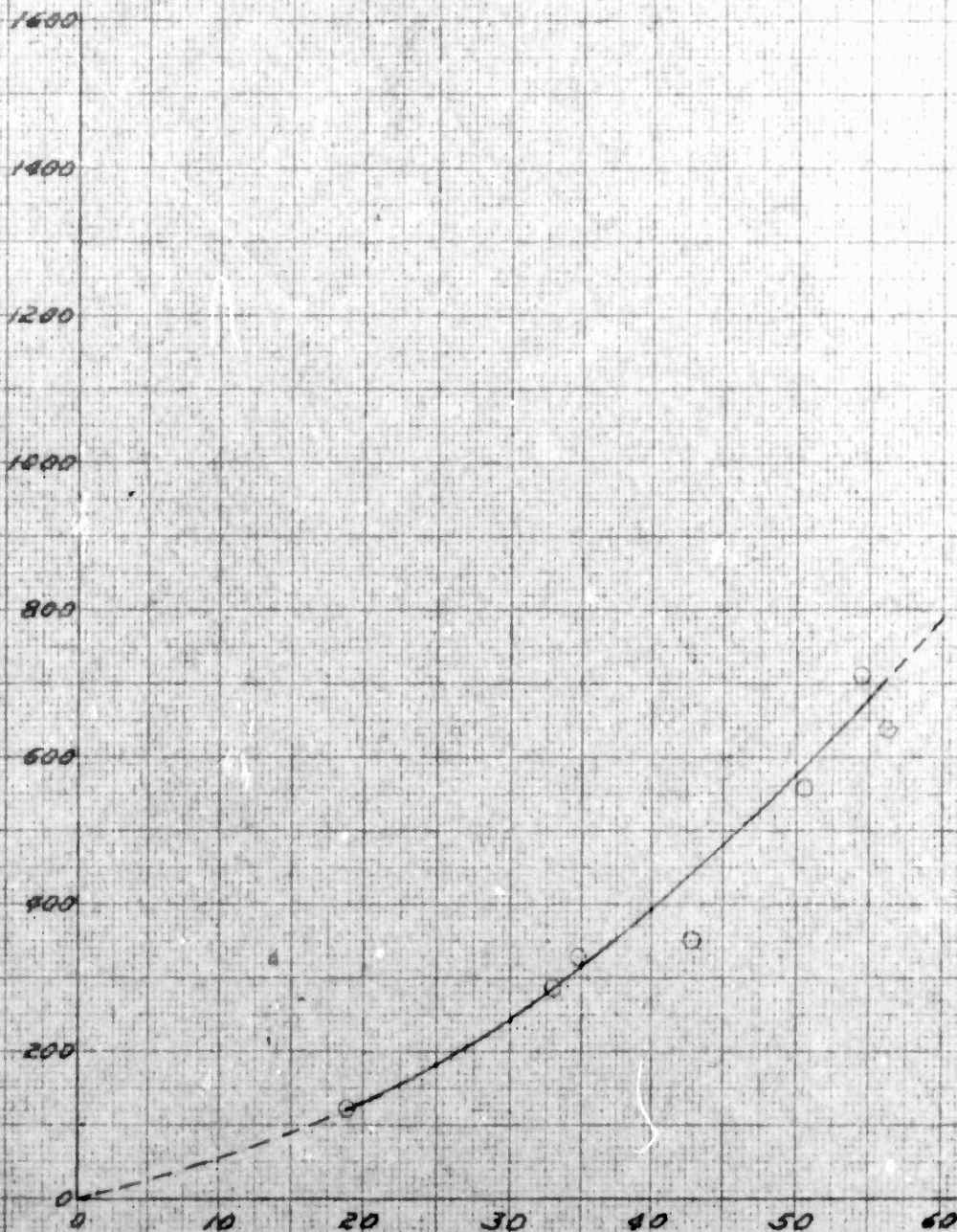


CLIMB-OUT TRUE AIRSPEED - KNOTS

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FIGURE No. 17
 TAKE-OFF DISTANCE REQUIRED
 TO CLEAR A 50' OBSTACLE
 OH-4A USA SING 62-4202
 CLEAN CONFIGURATION
 ROTOR RPM = 394
 GROSS WEIGHT = 2254 LB.
 FREE AIR TEMP = 14.4°C
 PRESSURE ALTITUDE = 4300 FT.
 WIND VELOCITY = 5.3 KNOTS
 $\Delta C_p = 4.63 \times 10^{-5}$
 TECHNIQUE: LEVEL ACCELERATION FROM A 2' HOVER

TOTAL DISTANCE REQUIRED TO CLEAR A 50 FOOT OBSTACLE - FEET



CLIMB-OUT TRUE AIRSPEED - KNOTS

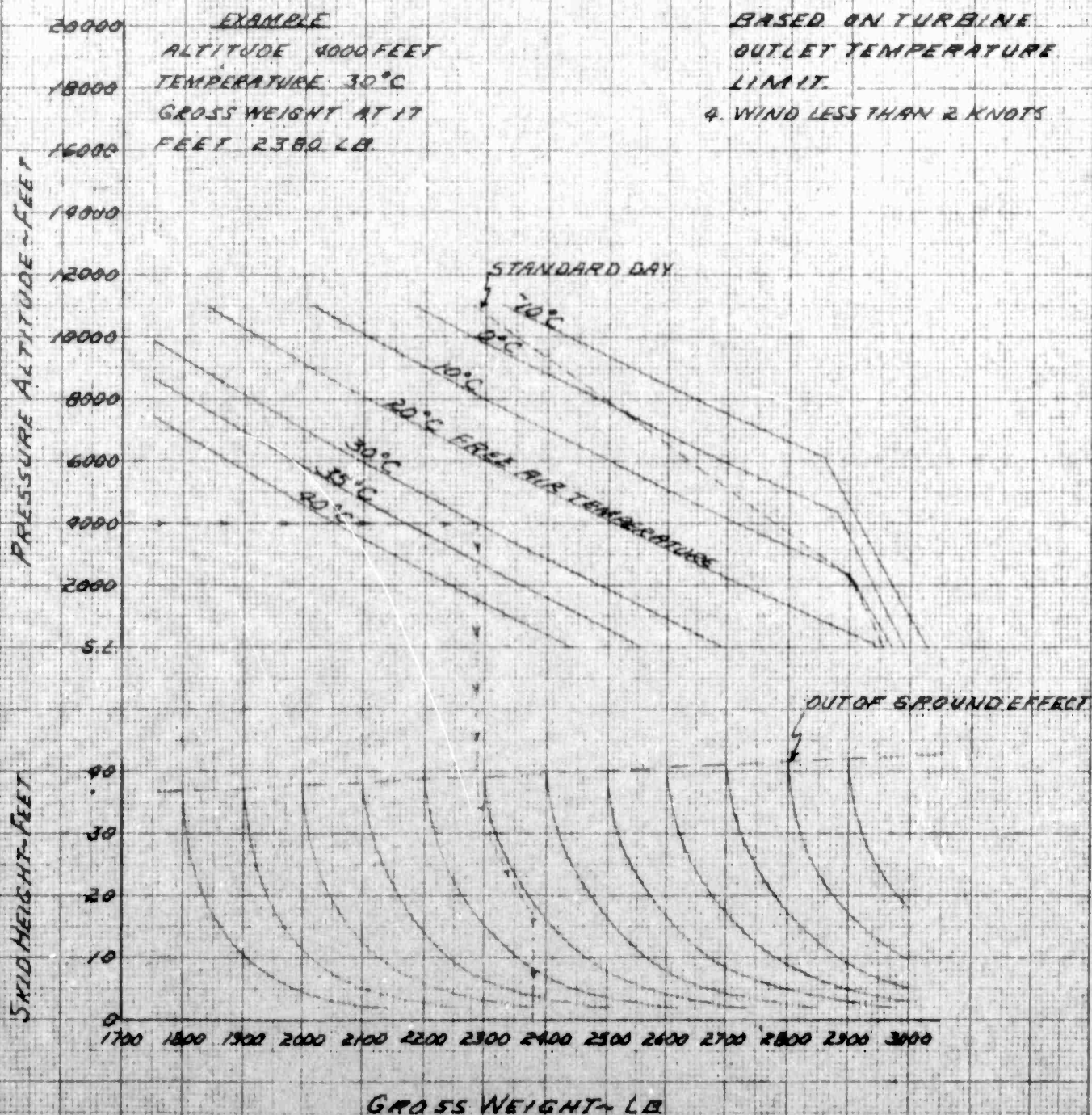
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FIGURE NO 18
SUMMARY HOVERING PERFORMANCE

OH-4A USA S/N 62-4202
ROTOR SPEED: 394 RPM
TAKE-OFF POWER
CLEAN CONFIGURATION

NOTES:

1. BASED ON ENGINE MODEL SPECIFICATION 580-E.
2. BELOW CRITICAL ALTITUDE BASED ON MAXIMUM ALLOWABLE TORQUE.
3. ABOVE CRITICAL ALTITUDE BASED ON TURBINE OUTLET TEMPERATURE LIMIT.
4. WIND LESS THAN 2 KNOTS

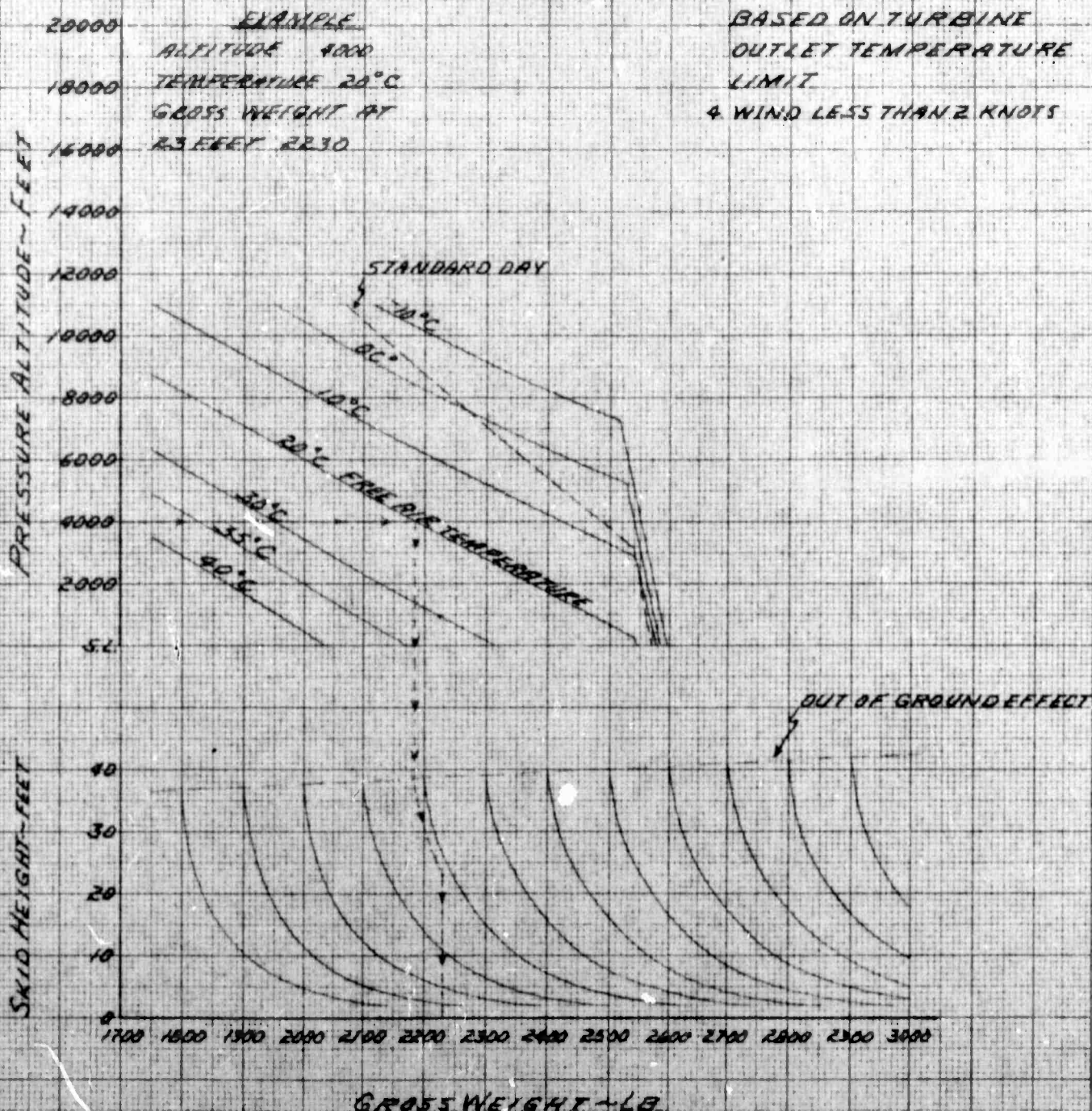


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FIGURE NO. 19
SUMMARY HOVERING PERFORMANCE
OH-4A USA S/N 62-4202
ROTOR SPEED - 394 RPM
MAXIMUM CONTINUOUS POWER
CLEAN CONFIGURATION

NOTES:

1. BASED ON ENGINE MODEL SPECIFICATION 580-E.
2. BELOW CRITICAL ALTITUDE BASED ON MAXIMUM ALLOWABLE TORQUE.
3. ABOVE CRITICAL ALTITUDE BASED ON TURBINE OUTLET TEMPERATURE LIMIT.
4. WIND LESS THAN 2 KNOTS

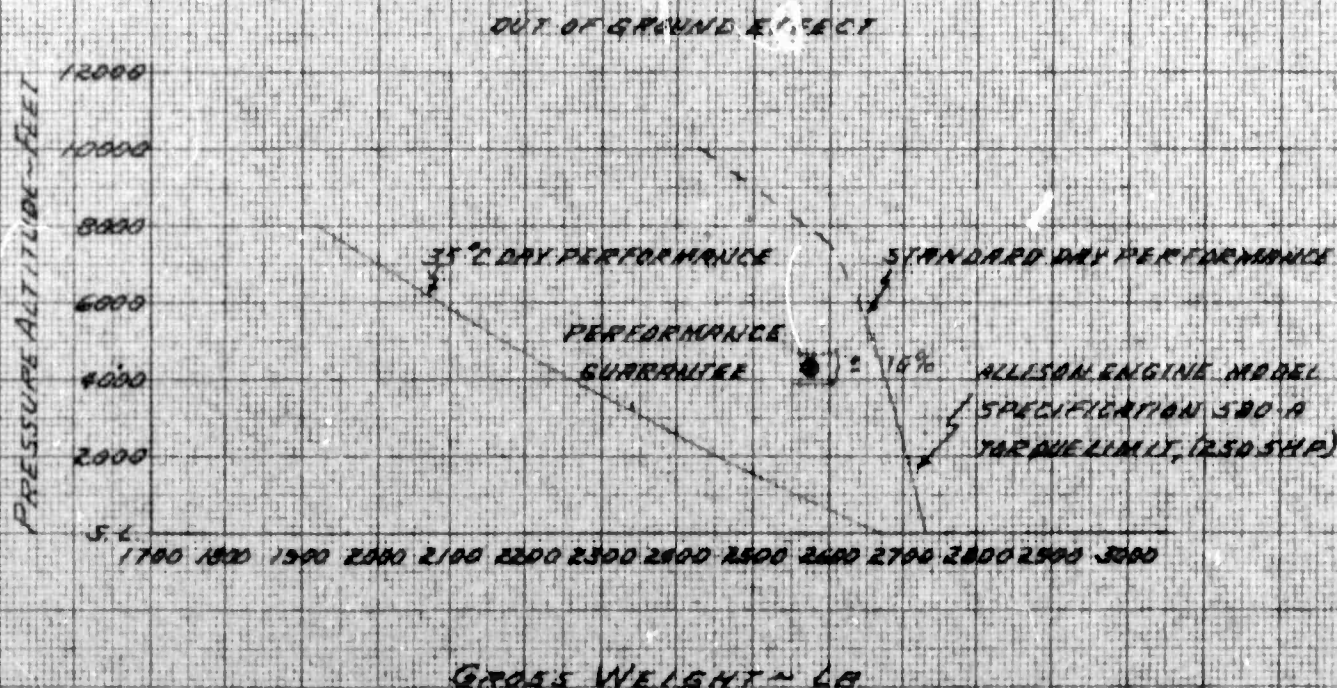


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FIGURE NO 20
SUMMARY HOVERING PERFORMANCE
OH-4A USA S/N 62-4202
ROTOR SPEED - 394 RPM
TAKE-OFF POWER
CLEAN CONFIGURATION

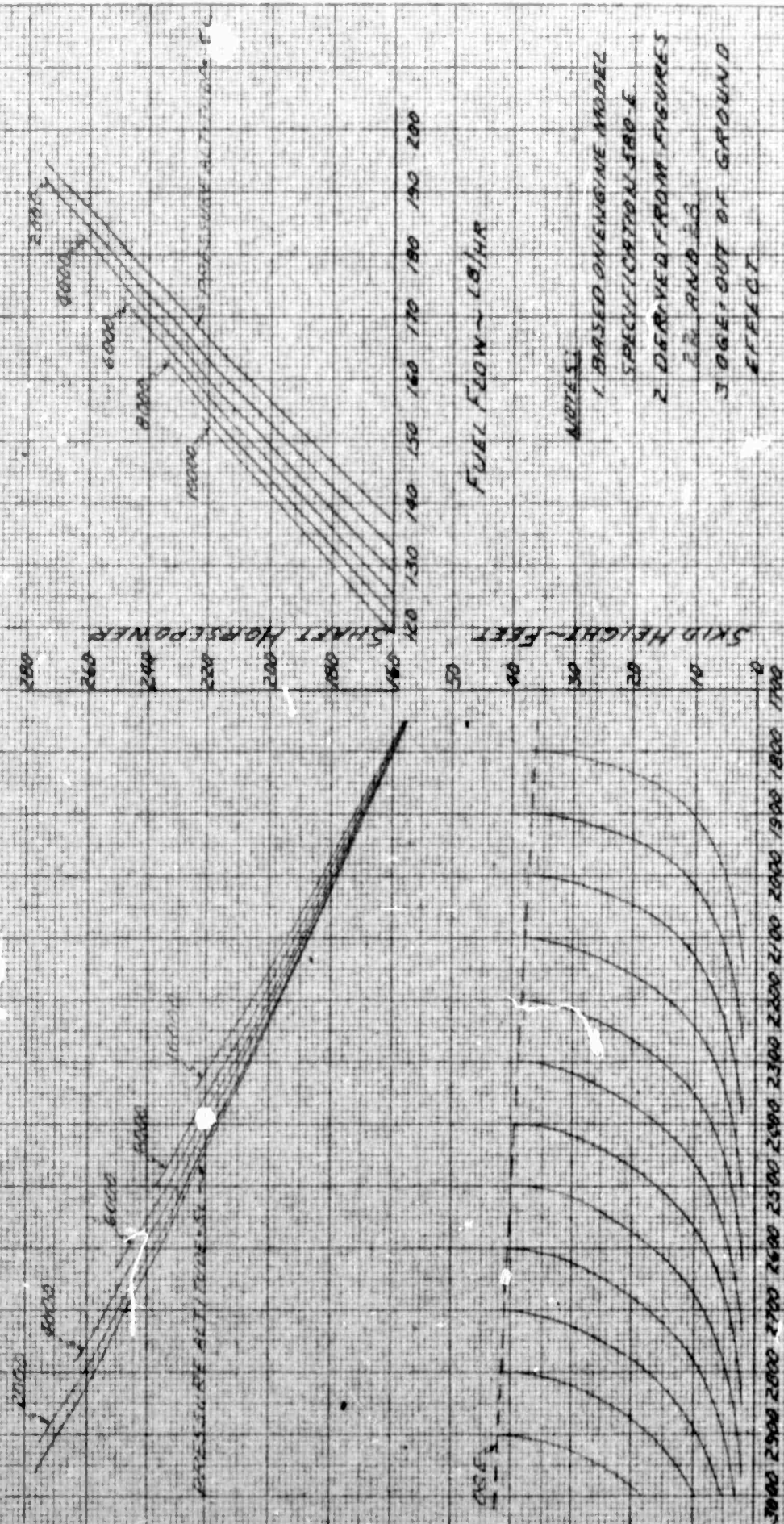
NOTES:

1. BASED ON ENGINE MODEL SPECIFICATION 580-A
2. BELOW CRITICAL ALTITUDE BASED ON MAXIMUM ALLOWABLE TORQUE
3. ABOVE CRITICAL ALTITUDE BASED ON TURBINE OUTLET TEMPERATURE LIMIT.



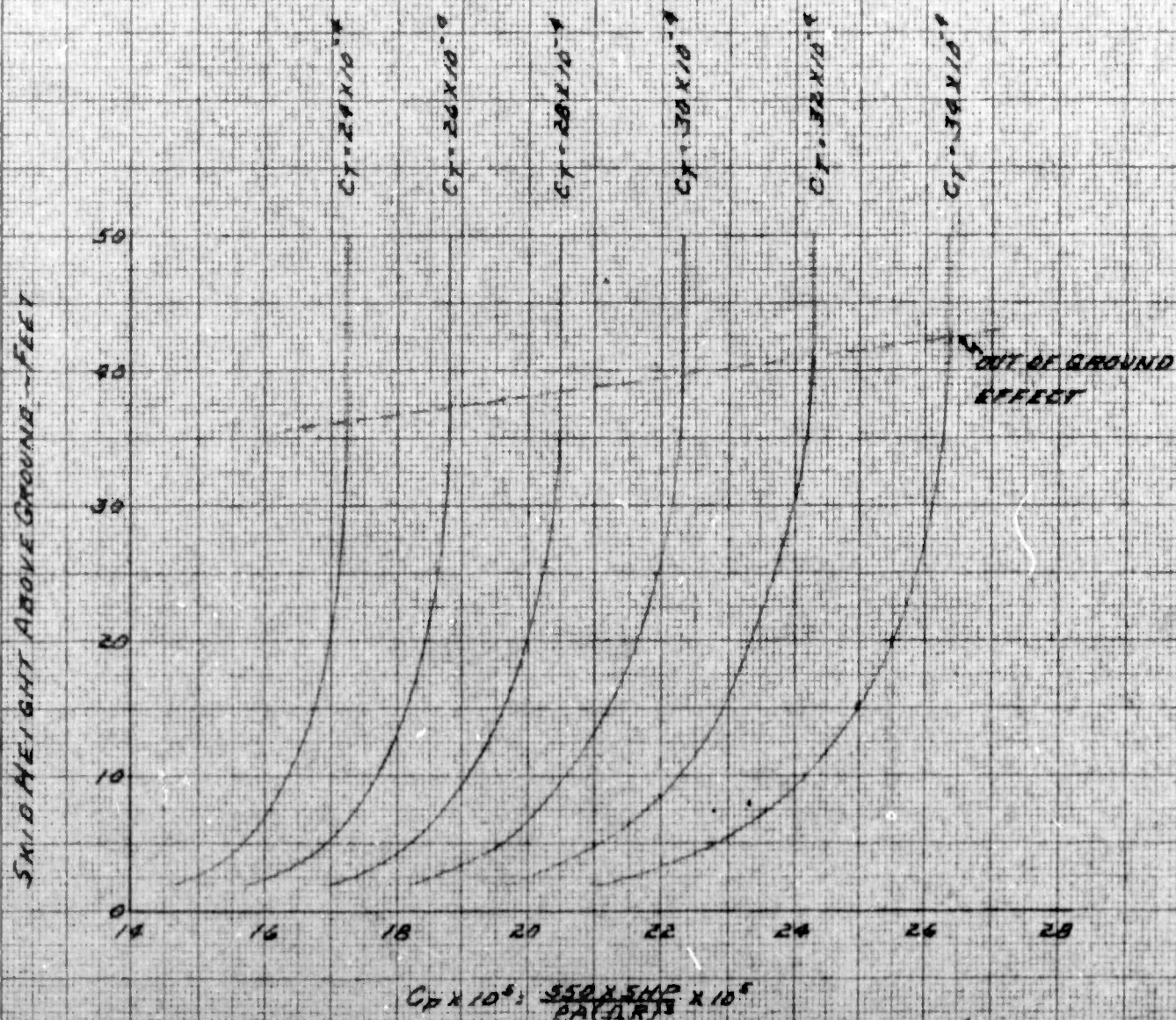
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FIGURE NO. 21
FUEL FLOW DURING HOVERING FLIGHT
OH-44 USA SN 62-4202
U.S. STANDARD DAY



ALBANY, N.Y. 12208
K. M. KELLEY & SONS CO.
10 KID TO THE CW. 3201-140

1. DERIVED FROM FAIRED CURVES ON FIGURE 29
2. VERTICAL DISTANCE FROM BOTTOM OF SKIDS TO CENTER OF ROTOR HUB
1.85 FEET



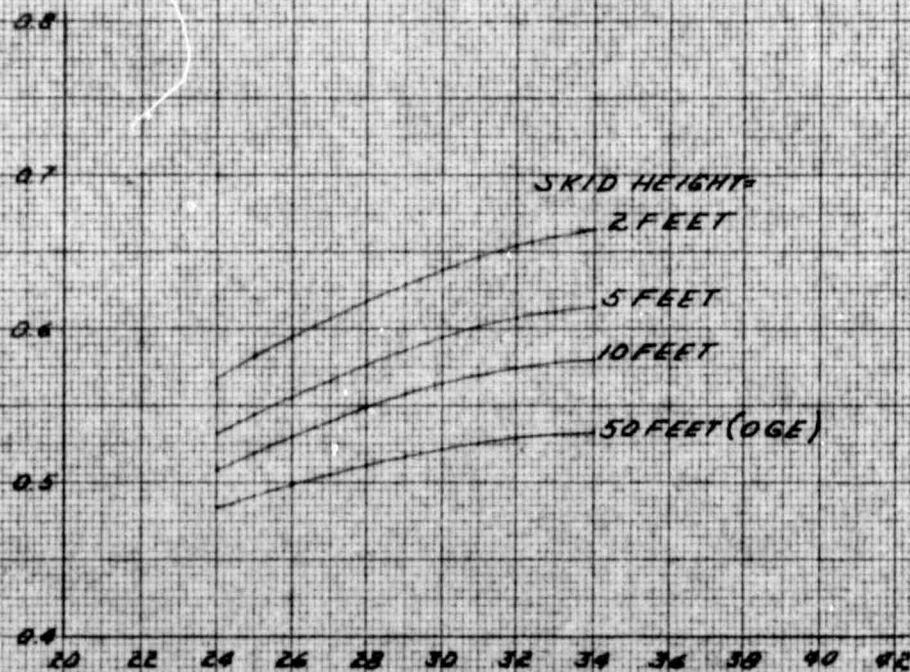
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FIGURE NO 23
NON-DIMENSIONAL HOVERING PERFORMANCE
OH-4A USA SIN 62-4202
CLEAN CONFIGURATION

NOTE:

- 1 DERIVED FROM FIGURES 25 THROUGH 28
- 2 VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB - 8.5 FEET
- 3 OGE - OUT OF GROUND EFFECT

FIGURE OF MERIT $M = 0.707 \times \frac{C_T}{C_D} \times \frac{V}{V_{ref}}$

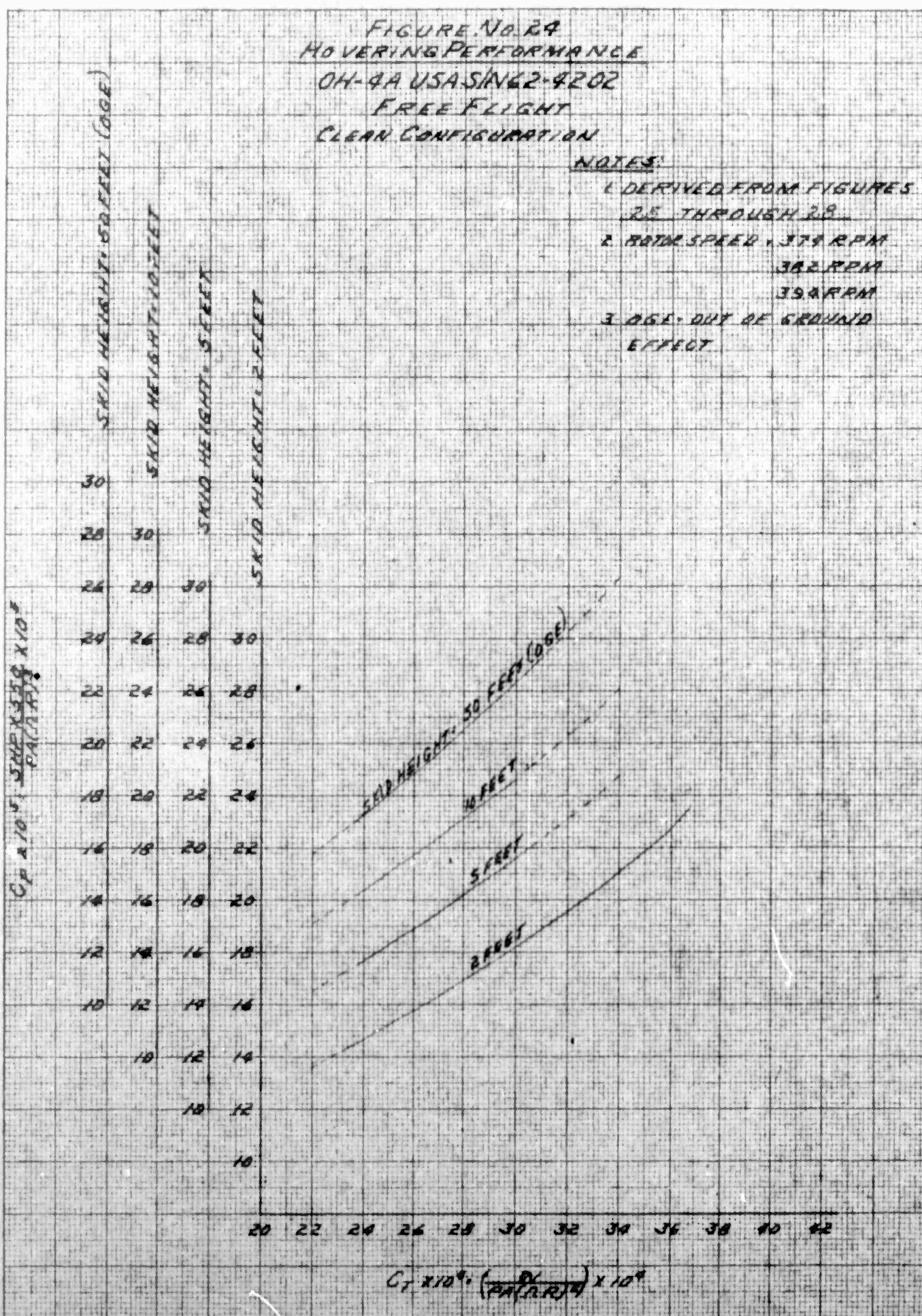


$C_T \times 10^4$ $\frac{W}{PA(1/2) \times 10^4}$

FIGURE NO. 29
HOVERING PERFORMANCE
OH-4A USASIN 62-4202
FREE FLIGHT
CLEAN CONFIGURATION

NOTES:

1. DERIVED FROM FIGURES 25 THROUGH 28.
2. ROTOR SPEED: 379 RPM
382 RPM
394 RPM
3. 0.61 DUE TO GROUND EFFECT.



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K-E

KENNAF & SORER CO.
10 X 10 TO THE CM. 320-140

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3

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 constitute fully legitimate reproduction

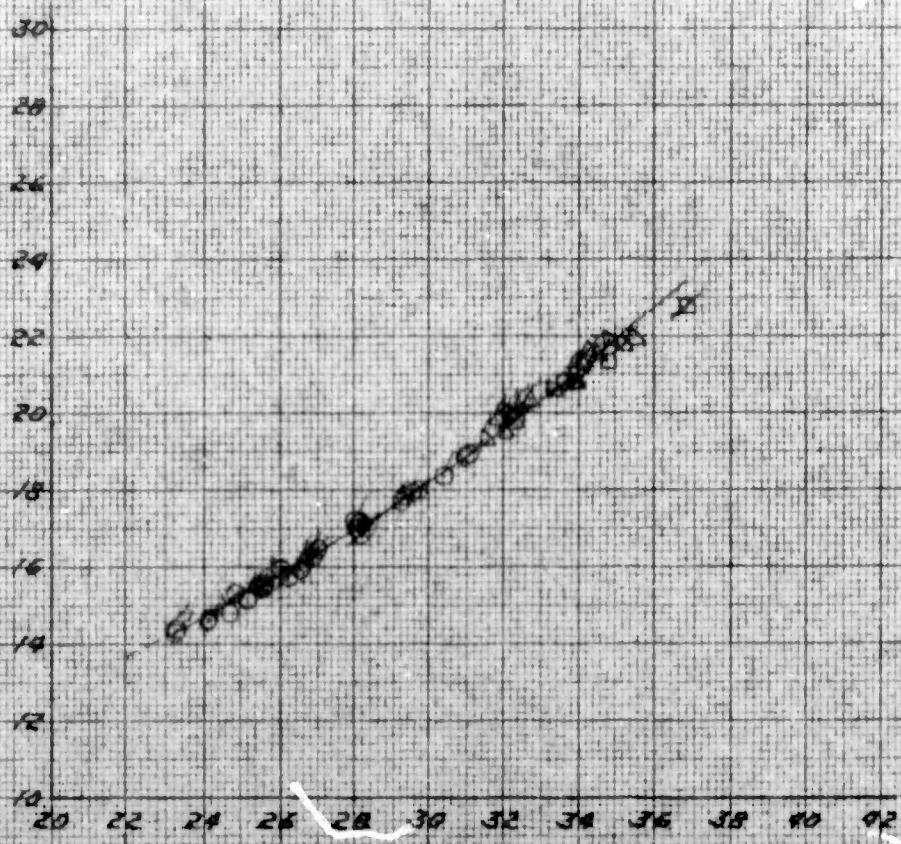
FIGURE NO 25
 HOVERING PERFORMANCE
 OH-4A USA SIN 62-4202
 FREE FLIGHT
 SKID HEIGHT 2 FEET
 CLEAN CONFIGURATION

NOTES

- 1. VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB 18.5 FEET
- 2. WIND LESS THAN 2 KNOTS

ROTOR SPEED-RPM			DENSITY ALTITUDE-FT
370	382	394	
0	0	0	100
0	0	0	5500
0	0	0	6000
0	0	0	10000

$C_p \times 10^5$ $C_{p0} \times 10^5$ $C_{p1} \times 10^5$



$C_T \times 10^4$ $C_{p0} \times 10^5$ $C_{p1} \times 10^5$

K.M. KENNEL & SONS CO. 3801-140

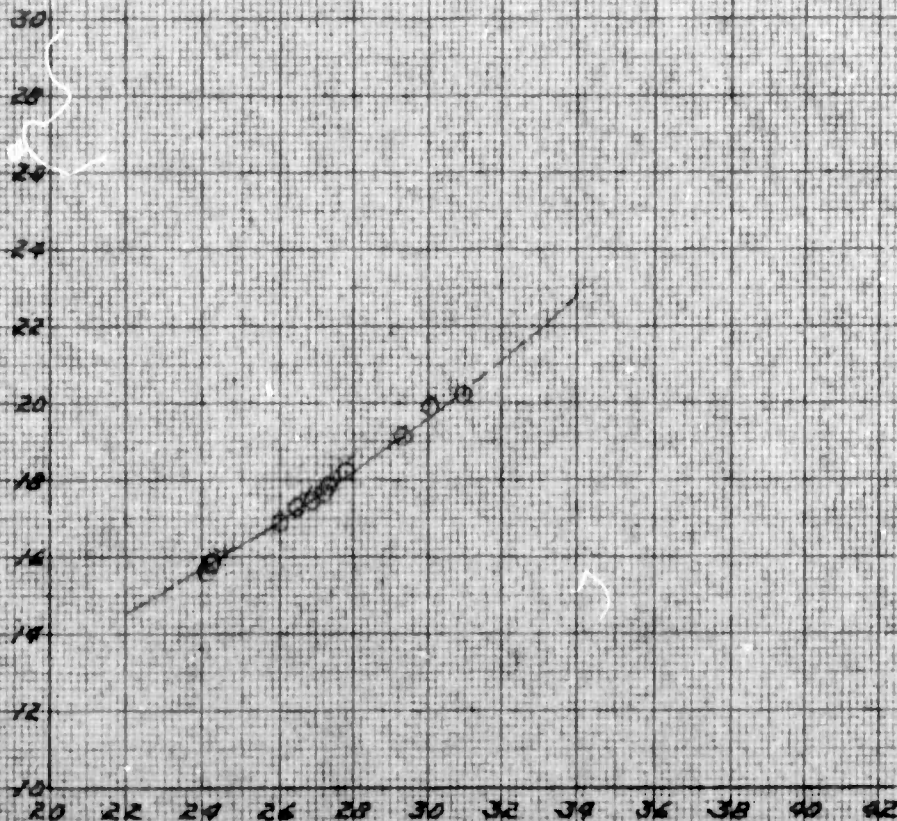
FIGURE NO. 26
HOVERING PERFORMANCE
OH-9A USA SIN 62-9202
FREE FLIGHT
SKID HEIGHT = 5 FEET
CLEAR CONFIGURATION

NOTES:

ROTOR SPEED-RPM DENSITY ALTITUDE-FT
374 382 394
0 0 0 140

VERTICAL DISTANCE FROM
BOTTOM OF SKID TO
CENTER OF ROTOR HUB
= 8.3 FEET
2. WIND LESS THAN 2 KNOTS

$C_T \times 10^4 = \frac{W}{P_{SL} \times 10^4}$



$C_T \times 10^4 = \frac{W}{P_{SL} \times 10^4}$

FIGURE NO 27
HOVERING PERFORMANCE
OH-4A USA SIN 62-4202
FREE FLIGHT
SKID HEIGHT-10 FEET
CLEAN CONFIGURATION

NOTES:

- 1 VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB 18.5 FEET
- 2 WIND LESS THAN 2 KNOTS

ROTOR SPEED-RPM	DENSITY ALTITUDE-FT
324	342
342	358
358	374
374	390
390	406
406	422
422	438
438	454
454	470
470	486
486	502
502	518
518	534
534	550
550	566
566	582
582	598
598	614
614	630
630	646
646	662
662	678
678	694
694	710
710	726
726	742
742	758
758	774
774	790
790	806
806	822
822	838
838	854
854	870
870	886
886	902
902	918
918	934
934	950
950	966
966	982
982	998
998	1014
1014	1030
1030	1046
1046	1062
1062	1078
1078	1094
1094	1110
1110	1126
1126	1142
1142	1158
1158	1174
1174	1190
1190	1206
1206	1222
1222	1238
1238	1254
1254	1270
1270	1286
1286	1302
1302	1318
1318	1334
1334	1350
1350	1366
1366	1382
1382	1398
1398	1414
1414	1430
1430	1446
1446	1462
1462	1478
1478	1494
1494	1510
1510	1526
1526	1542
1542	1558
1558	1574
1574	1590
1590	1606
1606	1622
1622	1638
1638	1654
1654	1670
1670	1686
1686	1702
1702	1718
1718	1734
1734	1750
1750	1766
1766	1782
1782	1798
1798	1814
1814	1830
1830	1846
1846	1862
1862	1878
1878	1894
1894	1910
1910	1926
1926	1942
1942	1958
1958	1974
1974	1990
1990	2006
2006	2022
2022	2038
2038	2054
2054	2070
2070	2086
2086	2102
2102	2118
2118	2134
2134	2150
2150	2166
2166	2182
2182	2198
2198	2214
2214	2230
2230	2246
2246	2262
2262	2278
2278	2294
2294	2310
2310	2326
2326	2342
2342	2358
2358	2374
2374	2390
2390	2406
2406	2422
2422	2438
2438	2454
2454	2470
2470	2486
2486	2502
2502	2518
2518	2534
2534	2550
2550	2566
2566	2582
2582	2598
2598	2614
2614	2630
2630	2646
2646	2662
2662	2678
2678	2694
2694	2710
2710	2726
2726	2742
2742	2758
2758	2774
2774	2790
2790	2806
2806	2822
2822	2838
2838	2854
2854	2870
2870	2886
2886	2902
2902	2918
2918	2934
2934	2950
2950	2966
2966	2982
2982	2998
2998	3014
3014	3030
3030	3046
3046	3062
3062	3078
3078	3094
3094	3110
3110	3126
3126	3142
3142	3158
3158	3174
3174	3190
3190	3206
3206	3222
3222	3238
3238	3254
3254	3270
3270	3286
3286	3302
3302	3318
3318	3334
3334	3350
3350	3366
3366	3382
3382	3398
3398	3414
3414	3430
3430	3446
3446	3462
3462	3478
3478	3494
3494	3510
3510	3526
3526	3542
3542	3558
3558	3574
3574	3590
3590	3606
3606	3622
3622	3638
3638	3654
3654	3670
3670	3686
3686	3702
3702	3718
3718	3734
3734	3750
3750	3766
3766	3782
3782	3798
3798	3814
3814	3830
3830	3846
3846	3862
3862	3878
3878	3894
3894	3910
3910	3926
3926	3942
3942	3958
3958	3974
3974	3990
3990	4006
4006	4022
4022	4038
4038	4054
4054	4070
4070	4086
4086	4102
4102	4118
4118	4134
4134	4150
4150	4166
4166	4182
4182	4198
4198	4214
4214	4230
4230	4246
4246	4262
4262	4278
4278	4294
4294	4310
4310	4326
4326	4342
4342	4358
4358	4374
4374	4390
4390	4406
4406	4422
4422	4438
4438	4454
4454	4470
4470	4486
4486	4502
4502	4518
4518	4534
4534	4550
4550	4566
4566	4582
4582	4598
4598	4614
4614	4630
4630	4646
4646	4662
4662	4678
4678	4694
4694	4710
4710	4726
4726	4742
4742	4758
4758	4774
4774	4790
4790	4806
4806	4822
4822	4838
4838	4854
4854	4870
4870	4886
4886	4902
4902	4918
4918	4934
4934	4950
4950	4966
4966	4982
4982	4998
4998	5014
5014	5030
5030	5046
5046	5062
5062	5078
5078	5094
5094	5110
5110	5126
5126	5142
5142	5158
5158	5174
5174	5190
5190	5206
5206	5222
5222	5238
5238	5254
5254	5270
5270	5286
5286	5302
5302	5318
5318	5334
5334	5350
5350	5366
5366	5382
5382	5398
5398	5414
5414	5430
5430	5446
5446	5462
5462	5478
5478	5494
5494	5510
5510	5526
5526	5542
5542	5558
5558	5574
5574	5590
5590	5606
5606	5622
5622	5638
5638	5654
5654	5670
5670	5686
5686	5702
5702	5718
5718	5734
5734	5750
5750	5766
5766	5782
5782	5798
5798	5814
5814	5830
5830	5846
5846	5862
5862	5878
5878	5894
5894	5910
5910	5926
5926	5942
5942	5958
5958	5974
5974	5990
5990	6006
6006	6022
6022	6038
6038	6054
6054	6070
6070	6086
6086	6102
6102	6118
6118	6134
6134	6150
6150	6166
6166	6182
6182	6198
6198	6214
6214	6230
6230	6246
6246	6262
6262	6278
6278	6294
6294	6310
6310	6326
6326	6342
6342	6358
6358	6374
6374	6390
6390	6406
6406	6422
6422	6438
6438	6454
6454	6470
6470	6486
6486	6502
6502	6518
6518	6534
6534	6550
6550	6566
6566	6582
6582	6598
6598	6614
6614	6630
6630	6646
6646	6662
6662	6678
6678	6694
6694	6710
6710	6726
6726	6742
6742	6758
6758	6774
6774	6790
6790	6806
6806	6822
6822	6838
6838	6854
6854	6870
6870	6886
6886	6902
6902	6918
6918	6934
6934	6950
6950	6966
6966	6982
6982	6998
6998	7014
7014	7030
7030	7046
7046	7062
7062	7078
7078	7094
7094	7110
7110	7126
7126	7142
7142	7158
7158	7174
7174	7190
7190	7206
7206	7222
7222	7238
7238	7254
7254	7270
7270	7286
7286	7302
7302	7318
7318	7334
7334	7350
7350	7366
7366	7382
7382	7398
7398	7414
7414	7430
7430	7446
7446	7462
7462	7478
7478	7494
7494	7510
7510	7526
7526	7542

FIGURE NO. 28
HOVERING PERFORMANCE
OH-4A USA SIN 62-4202
FREE FLIGHT
SKID HEIGHT - 50 FEET
CLEAN CONFIGURATION

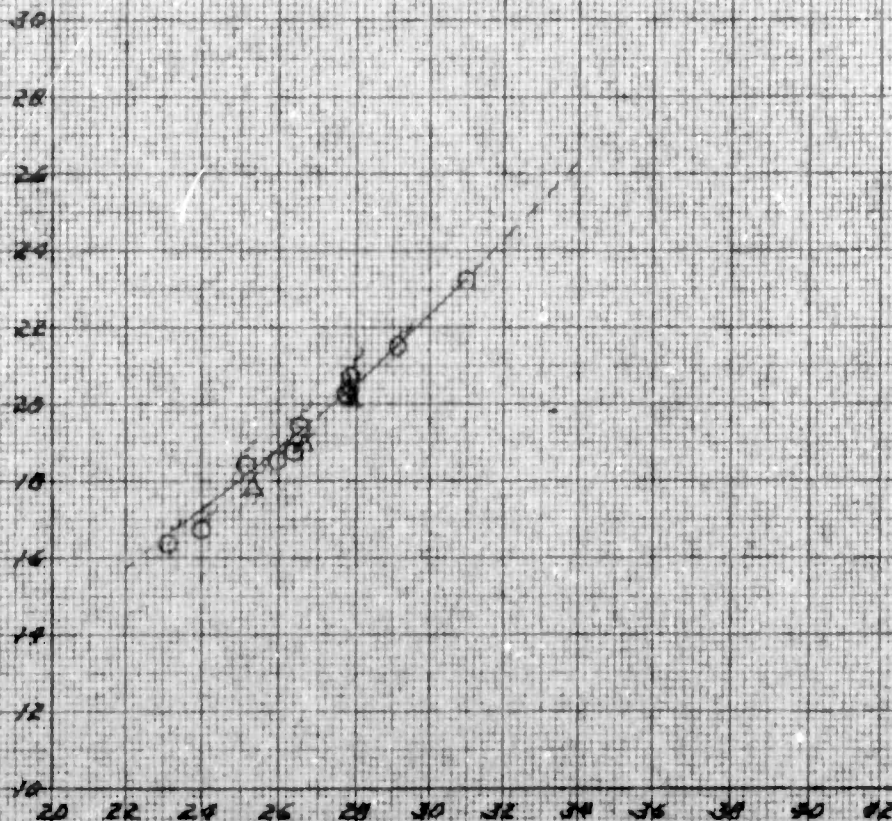
NOTES

1. VERTICAL DISTANCE FROM
BOTTOM OF SKID TO
CENTER OF ROTOR HUB
- 8.5 FEET.
2. WIND LESS THAN 2 KNOTS

ROTOR SPEED-RPM DENSITY ALTITUDE-FT.

374	382	394	
○	×	○	140
△	×	×	6750

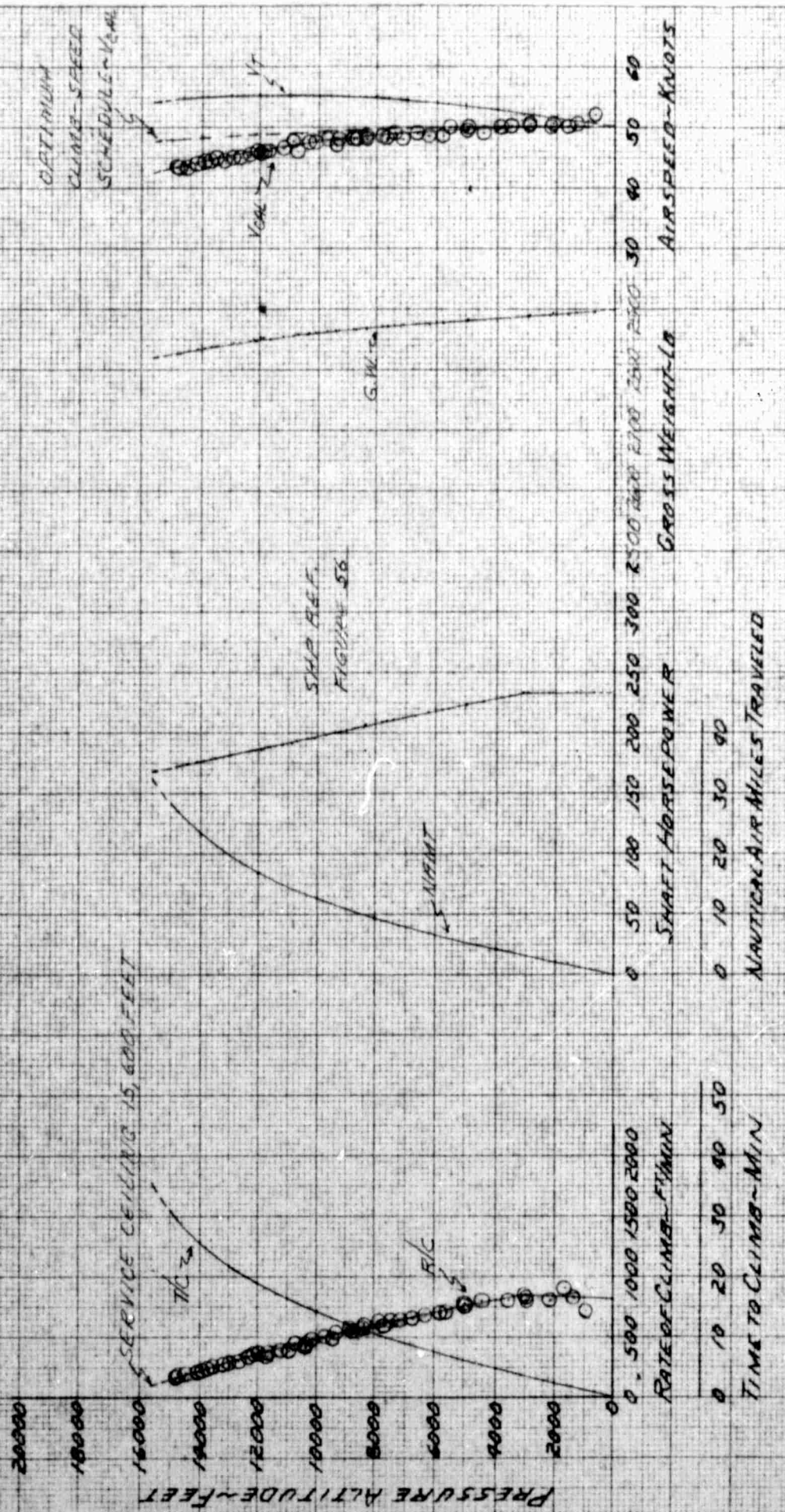
$C_P \times 10^3$ $\frac{W}{PAC(R)} \times 10^3$



$C_T \times 10^3 = \frac{W}{PAC(R)} \times 10^3$



FIGURE NO. 30
CLIMB PERFORMANCE
OH-4A USA SN 62-4202
MAXIMUM CONTINUOUS POWER
ROTOR SPEED 394 RPM
CG STATION 102.0 (N12)
U.S. STANDARD DAY
CLEAN CONFIGURATION



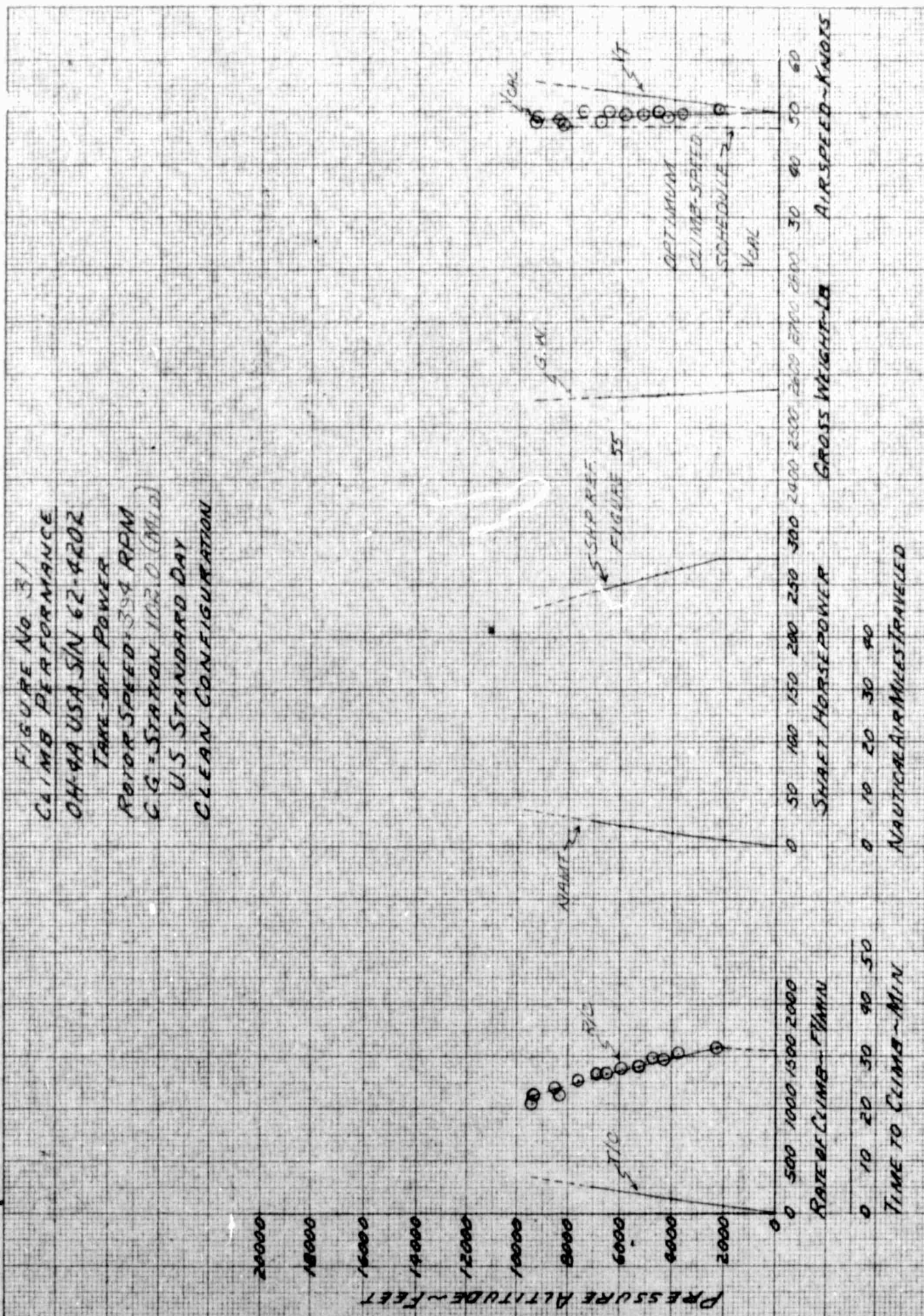


FIGURE NO 32
 CLIMB PERFORMANCE
 OH-4A USA SN 62-4202
 TAKE-OFF POWER
 ROTOR SPEED - 394 RPM
 CG - STATION 102.0 (HND)
 U.S. STANDARD DAY
 CLEAN CONFIGURATION

PRESSURE ALTITUDE - FEET

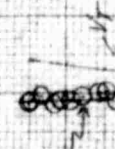


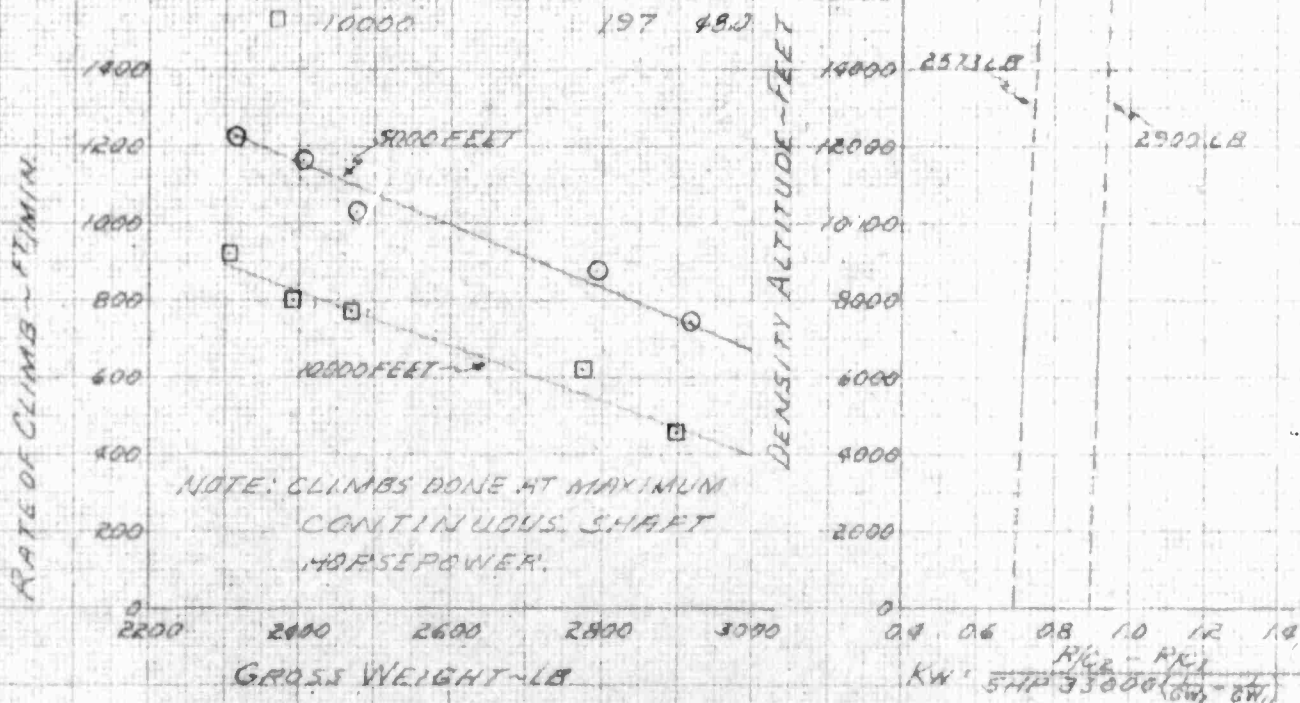
FIGURE NO. 33
VARIATION IN RATE OF CLIMB AS A FUNCTION OF
GROSS WEIGHT AND SHAFT HORSEPOWER

OH-4A USA SIN 62-4202
CLEAN CONFIGURATION
ROTOR SPEED: 394 RPM 18000

DENSITY ALTITUDE - FEET. SHP 5000 48.0 14000

○ 5000 197 48.0

□ 10000



DENSITY ALTITUDE - FEET

○ 5000

□ 10000

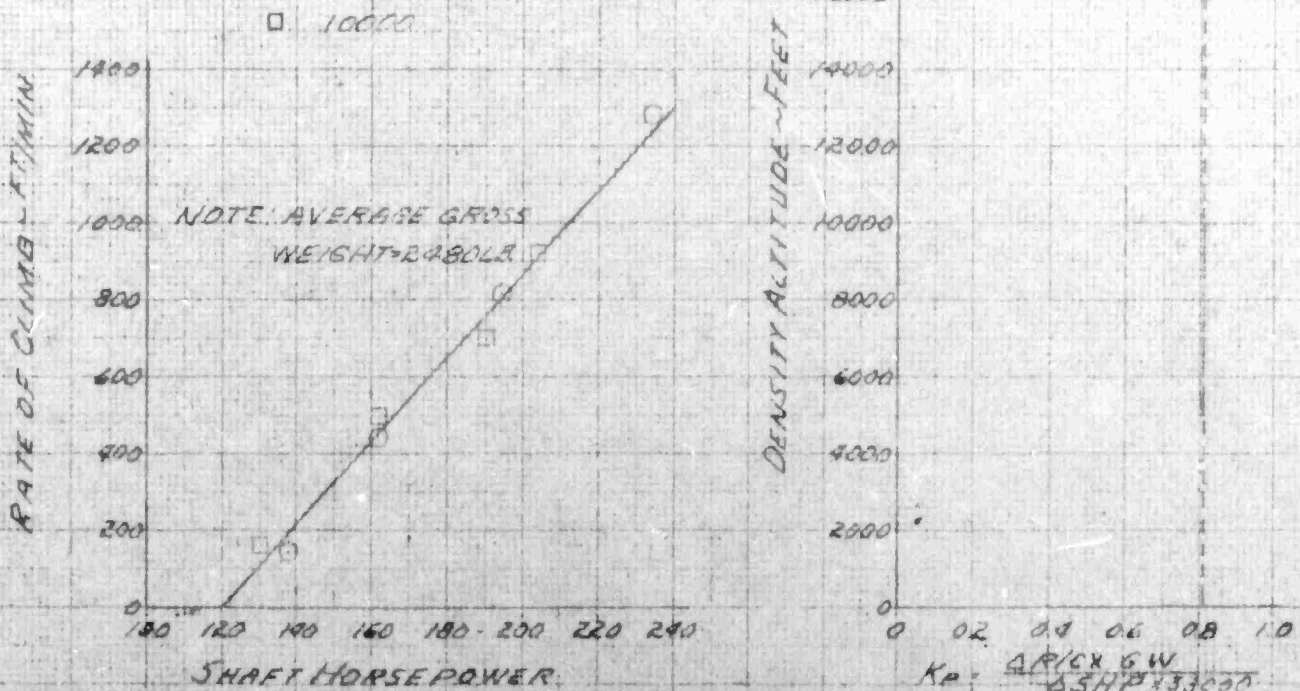
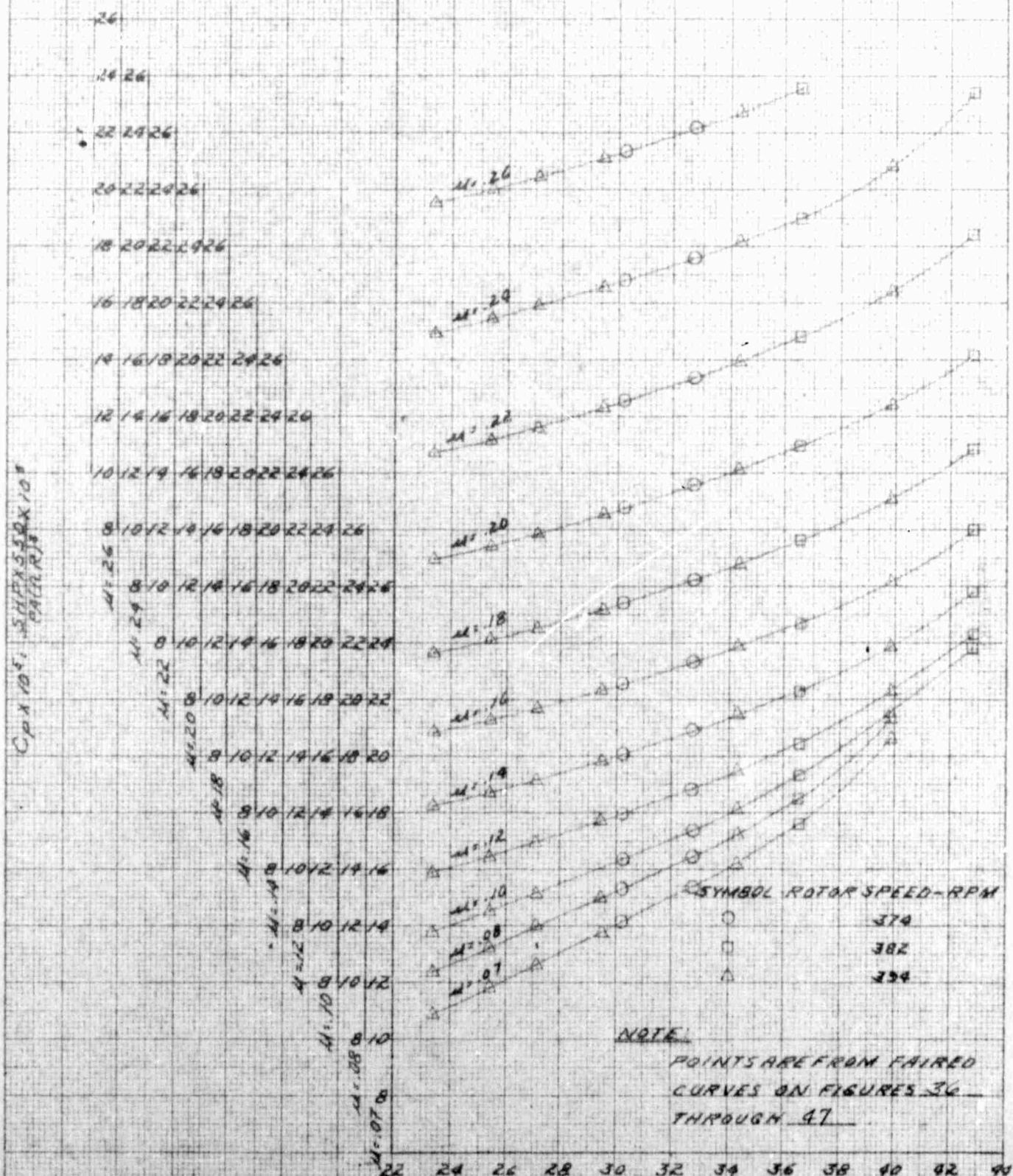


FIGURE NO. 34
LEVEL FLIGHT PERFORMANCE
OH-4A USA S/N 62-4202
C.G. STATION 1020 (MID)
CLEAN CONFIGURATION



NOTE

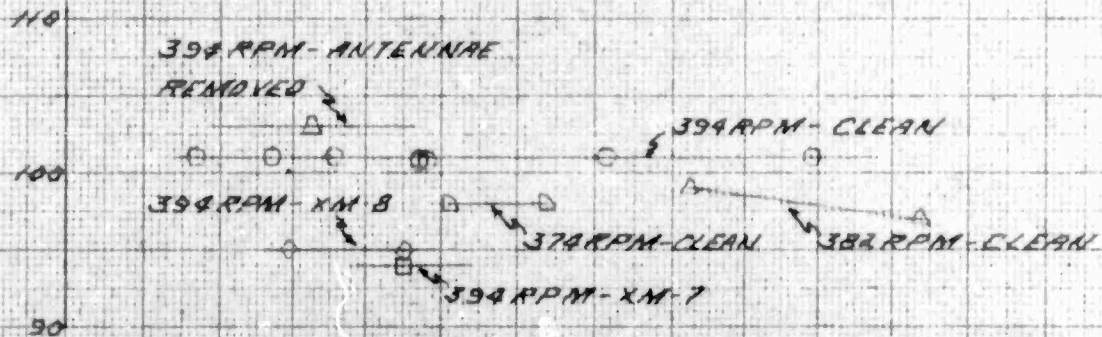
POINTS ARE FROM FAIRED CURVES ON FIGURES 36 THROUGH 47

$$C_T \times 10^4 = \frac{V}{PA(\Omega R)^2} \times 10^4$$

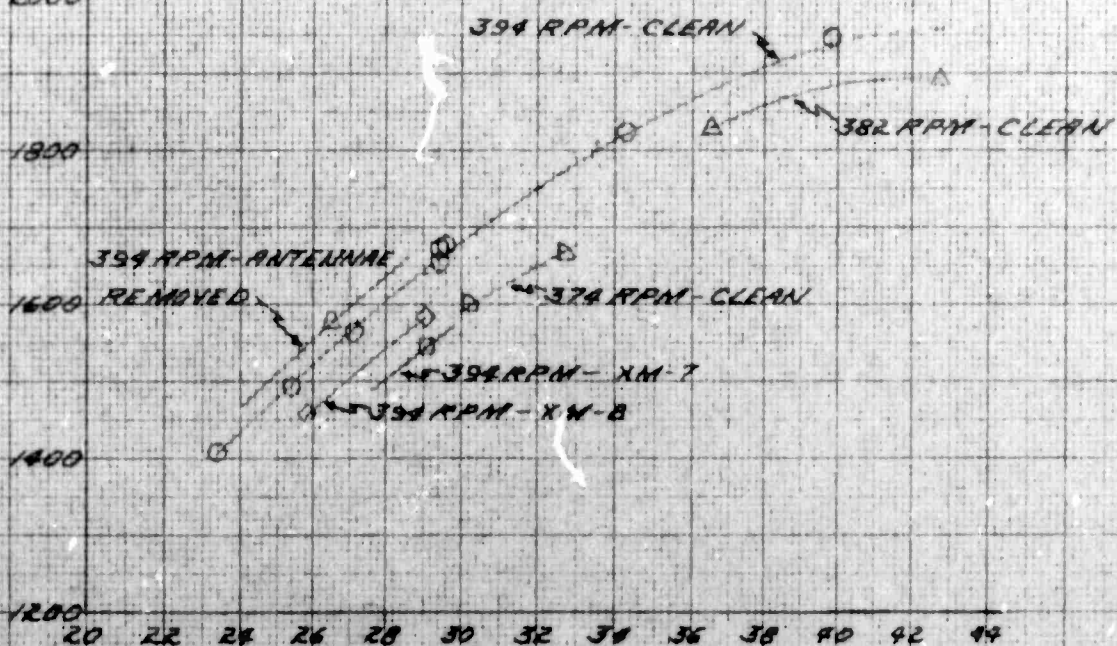
FIGURE NO 35
LEVEL FLIGHT RANGE SUMMARY
DH 4A USA SN 62-4202

SYMBOL	ROTOR SPEED-RPM	CONFIGURATION
○	394	CLEAN
□	394	XM-7
○	394	XM-8
△	382	CLEAN
◇	374	CLEAN
◇	394	ANTENNAE REMOVED

CRUISE V_{true} - KNOTS

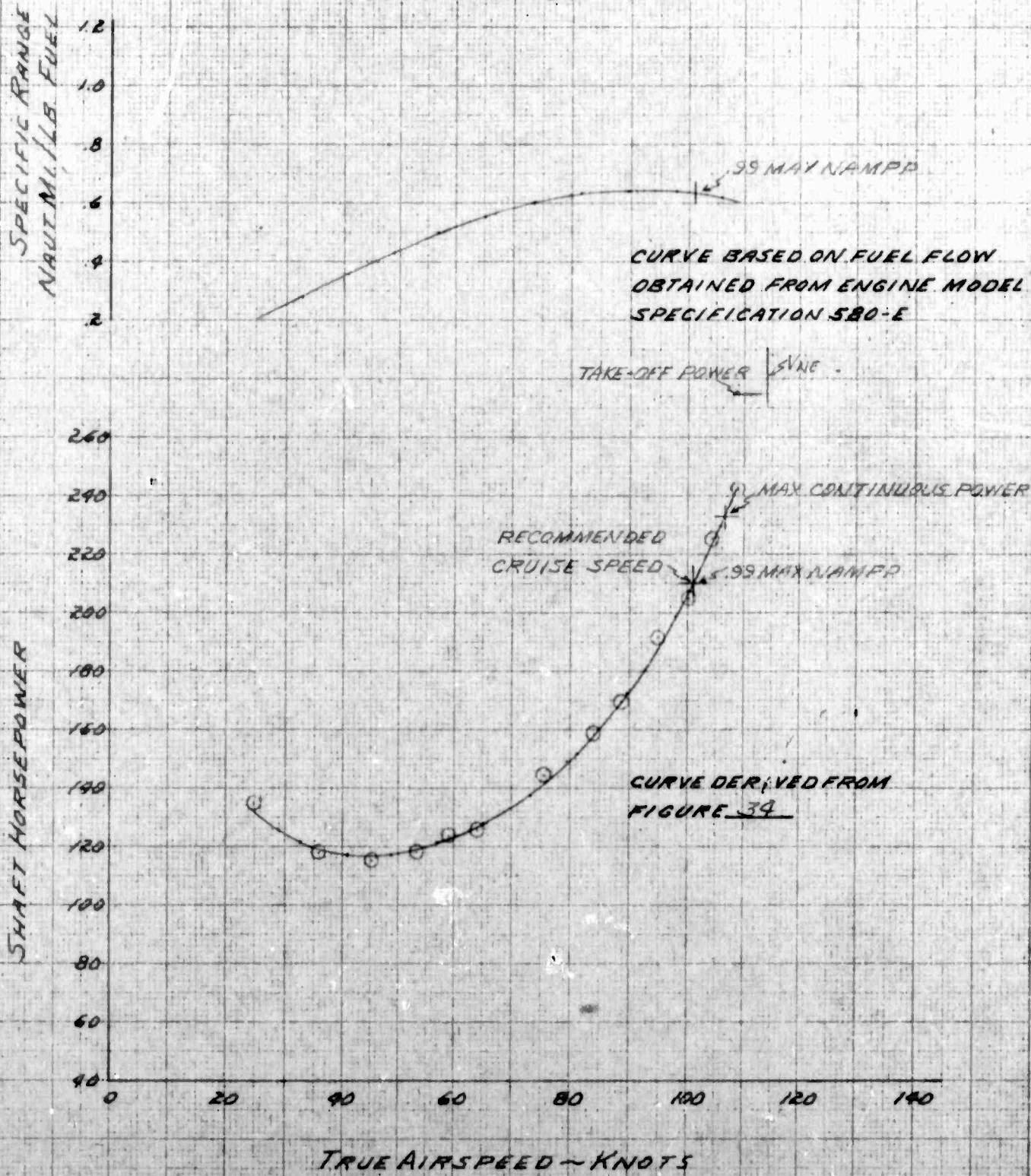


RANGE FACTOR - 99 NAMPP X WEIGHT



$$G_T \times 10^4 = \frac{W}{P_A(R.P.)^2} \times 10^4$$

FIGURE NO. 36
 LEVEL FLIGHT PERFORMANCE
 OH-9A USA S/N 62-4202
 GROSS WEIGHT - 2213 LB
 DENSITY ALTITUDE - 1225 FT.
 ROTOR SPEED - 394 RPM
 C.G. - STATION 102.1 (MID)
 $C_T = 23.39 \times 10^{-4}$
 CLEAN CONFIGURATION



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 PROJECT & RESEARCH
 10 X 10 TO THE CM
 3201 140

FIGURE NO. 37
LEVEL FLIGHT PERFORMANCE

OH-4A USA S/N 62-4202
GROSS WEIGHT - 2431 LB.
DENSITY ALTITUDE - 825 FT.
ROTOR SPEED - 394 RPM
C.G. - STATION 102.15 (MID)
 $C_T = 25.40 \times 10^{-4}$
CLEAN CONFIGURATION

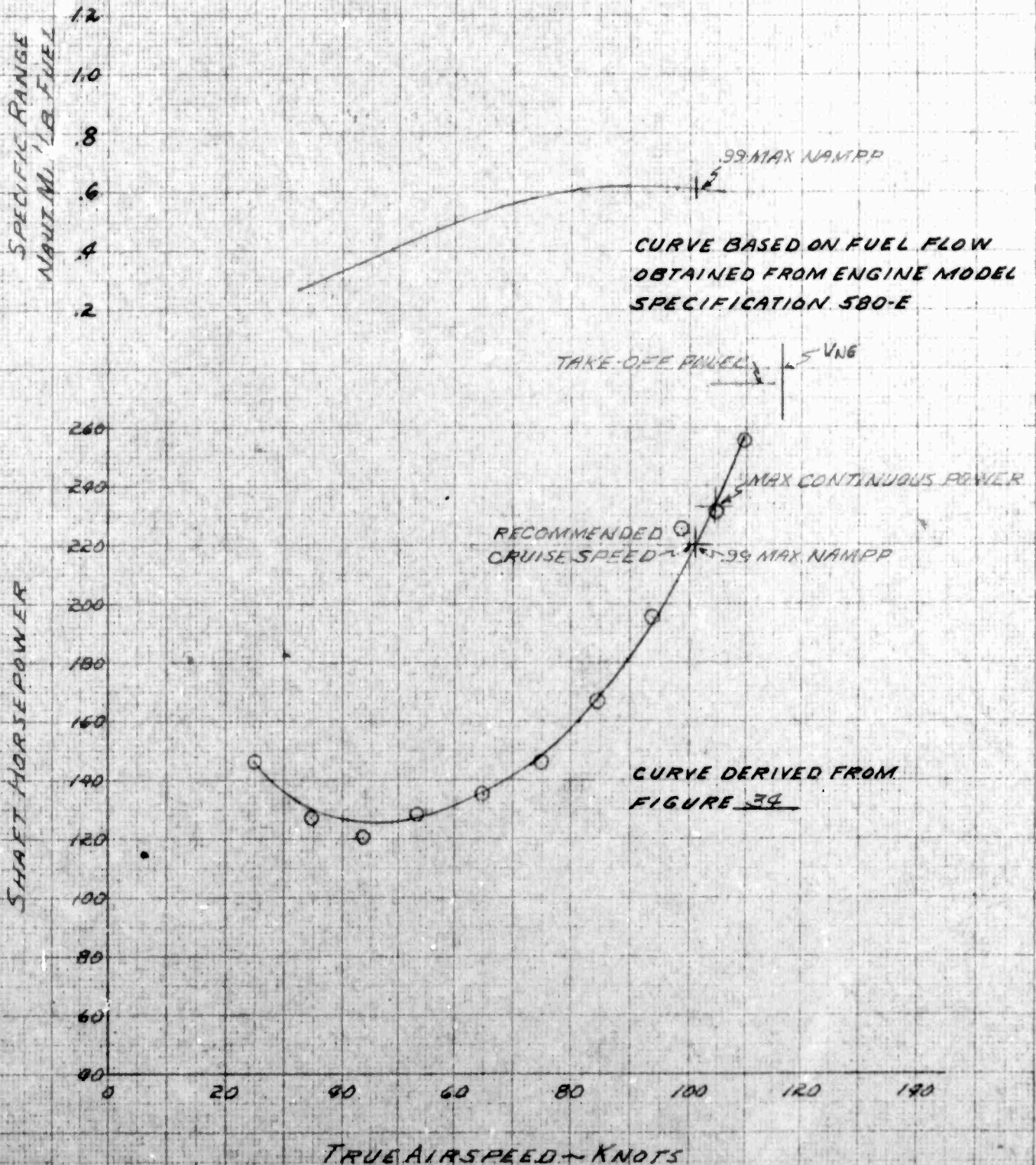


FIGURE NO. 3B
LEVEL FLIGHT PERFORMANCE

OH-4A USA S/N 62-4202
GROSS WEIGHT: 2232 LB.
DENSITY ALTITUDE: 5870 FT
ROTOR SPEED: 394 RPM
C.G. - STATION 102.2 (MID)
 $C_T = 2.712 \times 10^{-4}$
CLEAN CONFIGURATION

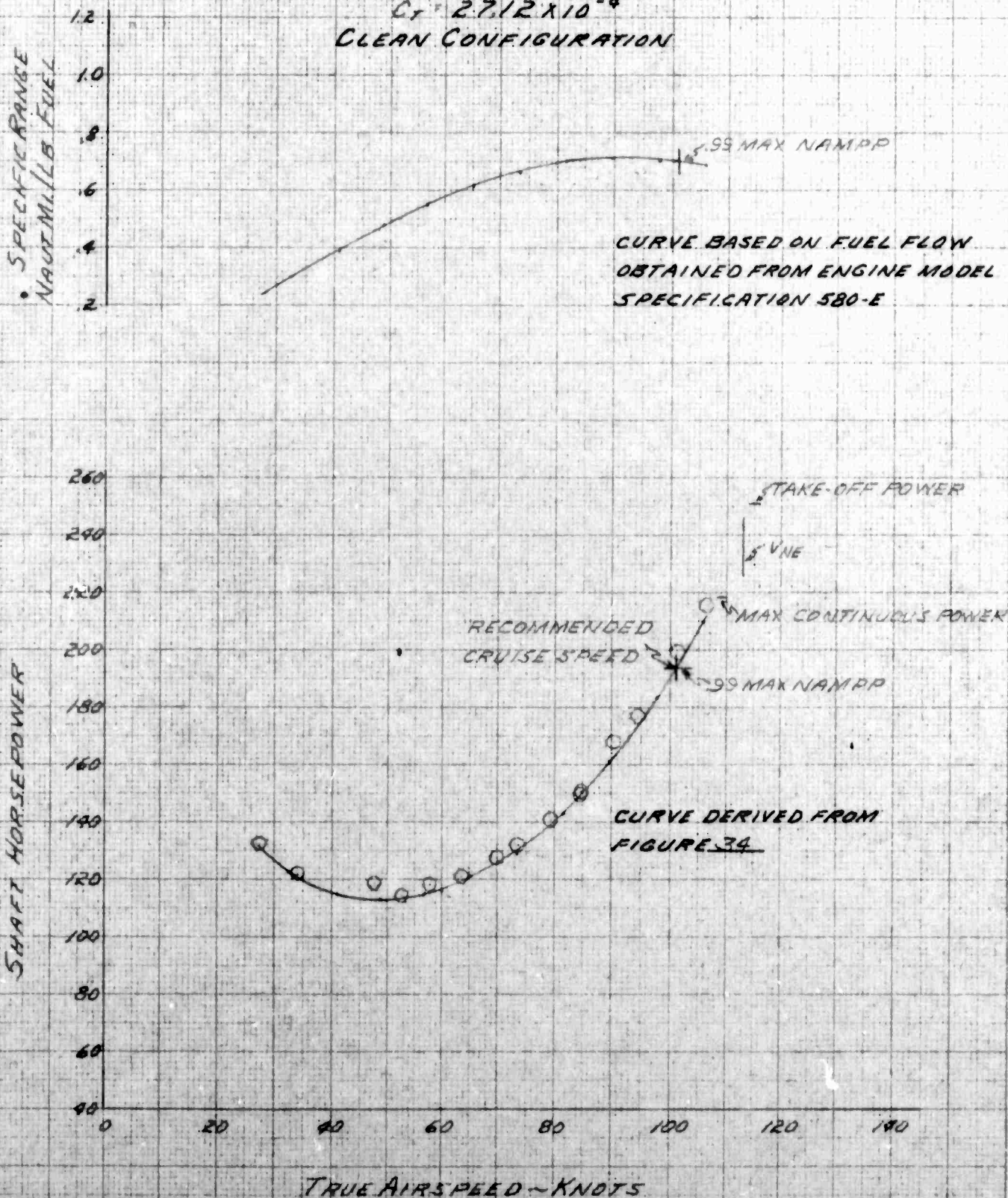


FIGURE NO 39
LEVEL FLIGHT PERFORMANCE

OH-4A USA SIN 62-4202
GROSS WEIGHT - 2435 LB
DENSITY ALTITUDE - 5525 FT.
ROTOR SPEED - 394 RPM
C.G. - STATION 99.3 (FWD)
 $G_T = 29.42 \times 10^{-9}$
CLEAN CONFIGURATION

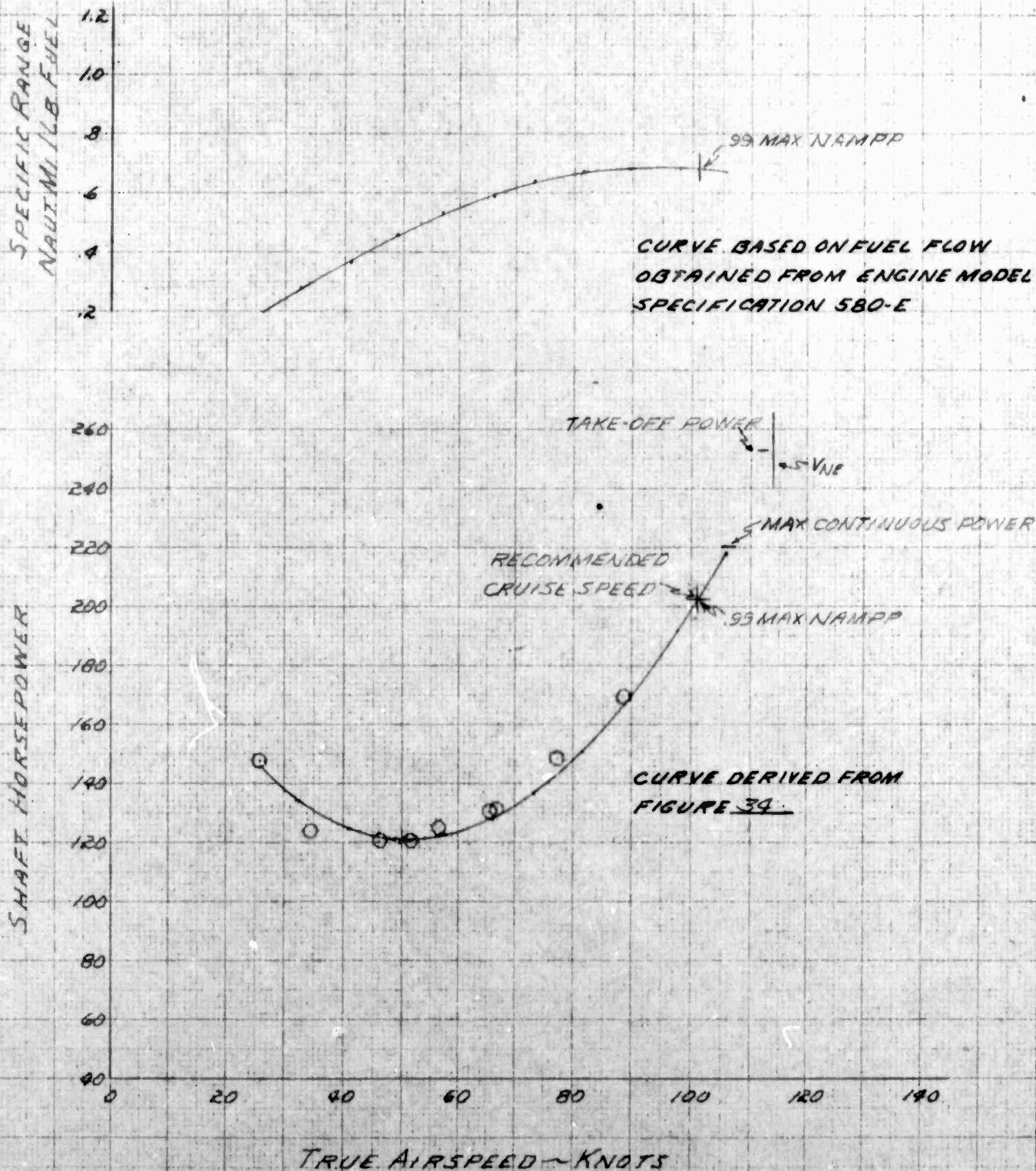


FIGURE NO. 40
LEVEL FLIGHT PERFORMANCE

OH-4A USA SIN 62-4202
GROSS WEIGHT: 2389 LB.
DENSITY ALTITUDE: 6350 FT
ROTOR SPEED: 394 RPM
C.G. - STATION 106.3 (AFT)
 $C_T = 29.45 \times 10^{-4}$
CLEAN CONFIGURATION

SPECIFIC RANGE
NAUT MI / LB FUEL

1.2
1.0
.8
.6
.4
.2

.99 MAX NAMPP

CURVE BASED ON FUEL FLOW
OBTAINED FROM ENGINE MODEL
SPECIFICATION 580-E

SHAFT HORSEPOWER

260
240
220
200
180
160
140
120
100
80
60
40
0

TAKE-OFF POWER

SVNE

MAX CONTINUOUS POWER

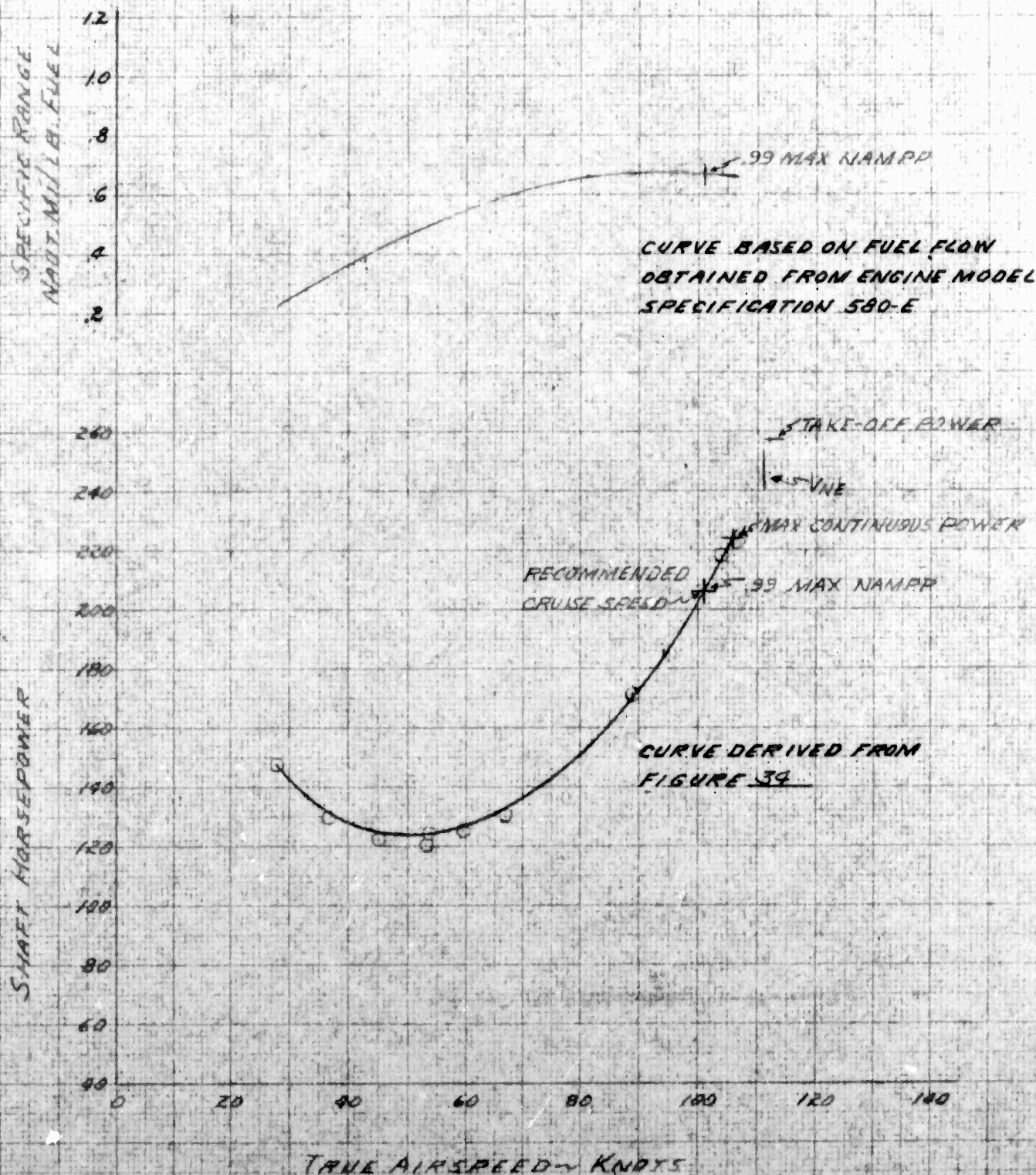
RECOMMENDED
CRUISE SPEED

.99 MAX NAMPP

CURVE DERIVED FROM
FIGURE 39

TRUE AIRSPEED - KNOTS

FIGURE NO 41
 LEVEL FLIGHT PERFORMANCE
 OH-4A USA S/N 62-4202
 GROSS WEIGHT 2499 LB
 DENSITY ALTITUDE 4905 FT
 ROTOR SPEED 394 RPM
 CG STATION 102.4 (MID)
 $C_T = 2952 \times 10^{-6}$
 CLEAN CONFIGURATION



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FIGURE NO. 42
LEVEL FLIGHT PERFORMANCE

OH-4A USA S/N 62-4202

GROSS WEIGHT: 2245 LB.

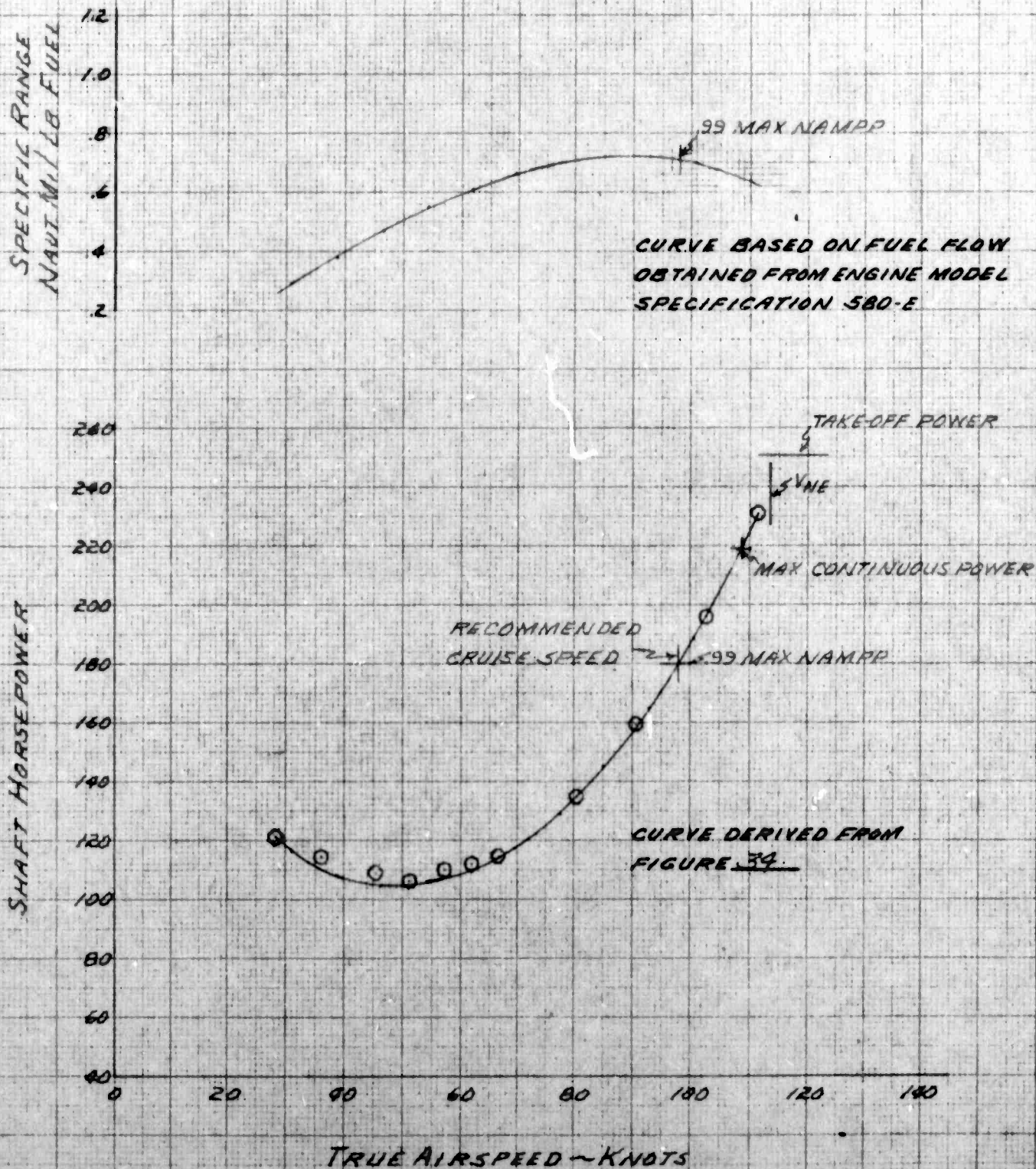
DENSITY ALTITUDE: 5805 FT.

ROTOR SPEED: 374 RPM

C.G. STATION 102.3 (MID)

$C_T = 3022 \times 10^{-4}$

CLEAN CONFIGURATION



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FIGURE NO. 43
LEVEL FLIGHT PERFORMANCE

OH-4A USA SN 62-4202
GROSS WEIGHT - 2806 LB.
DENSITY ALTITUDE - 1030 FT.
ROTOR SPEED - 374 RPM
C.G. - STATION 101.8 (MID)
 $C_T = 32.71 \times 10^{-6}$
CLEAN CONFIGURATION

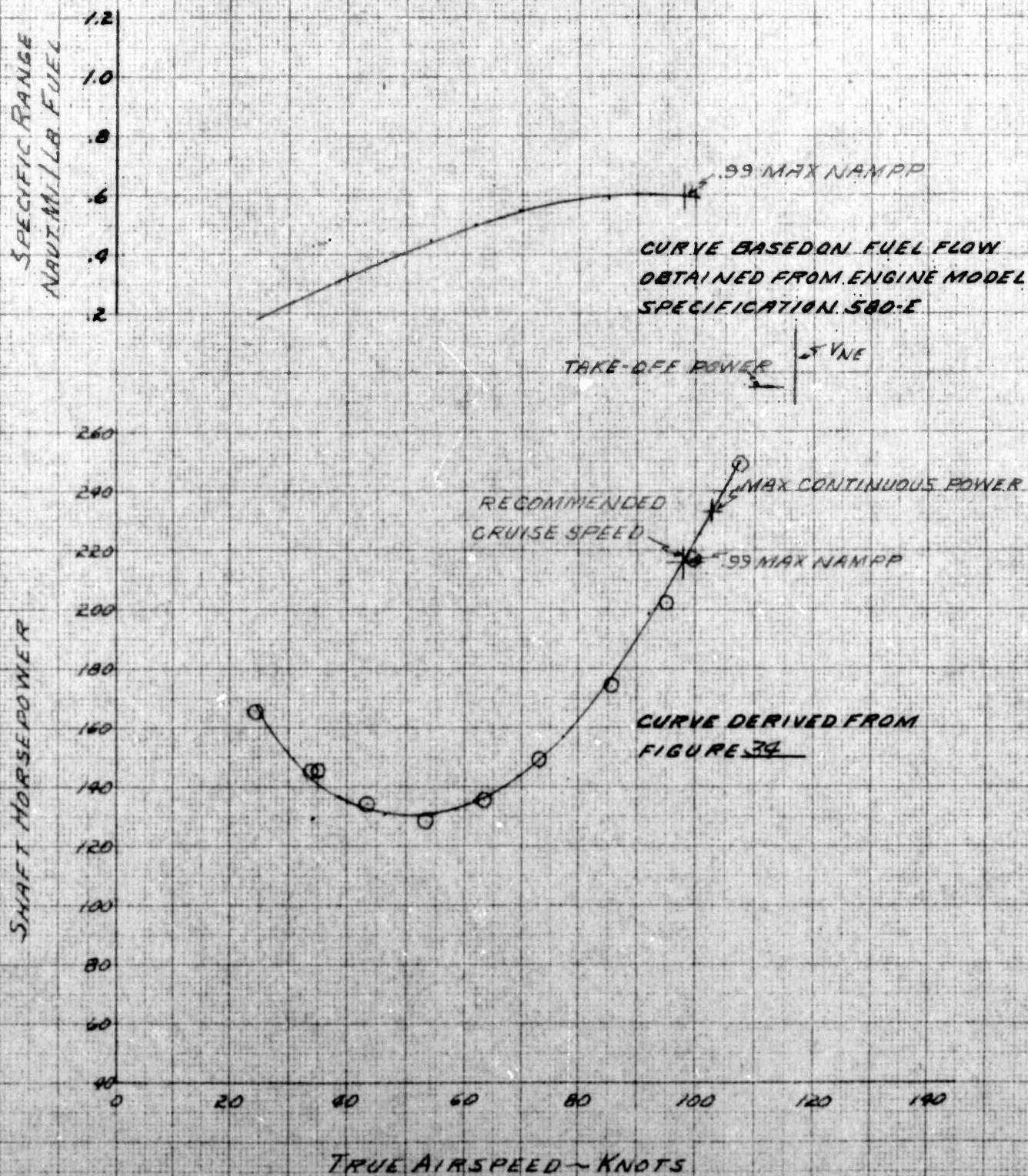
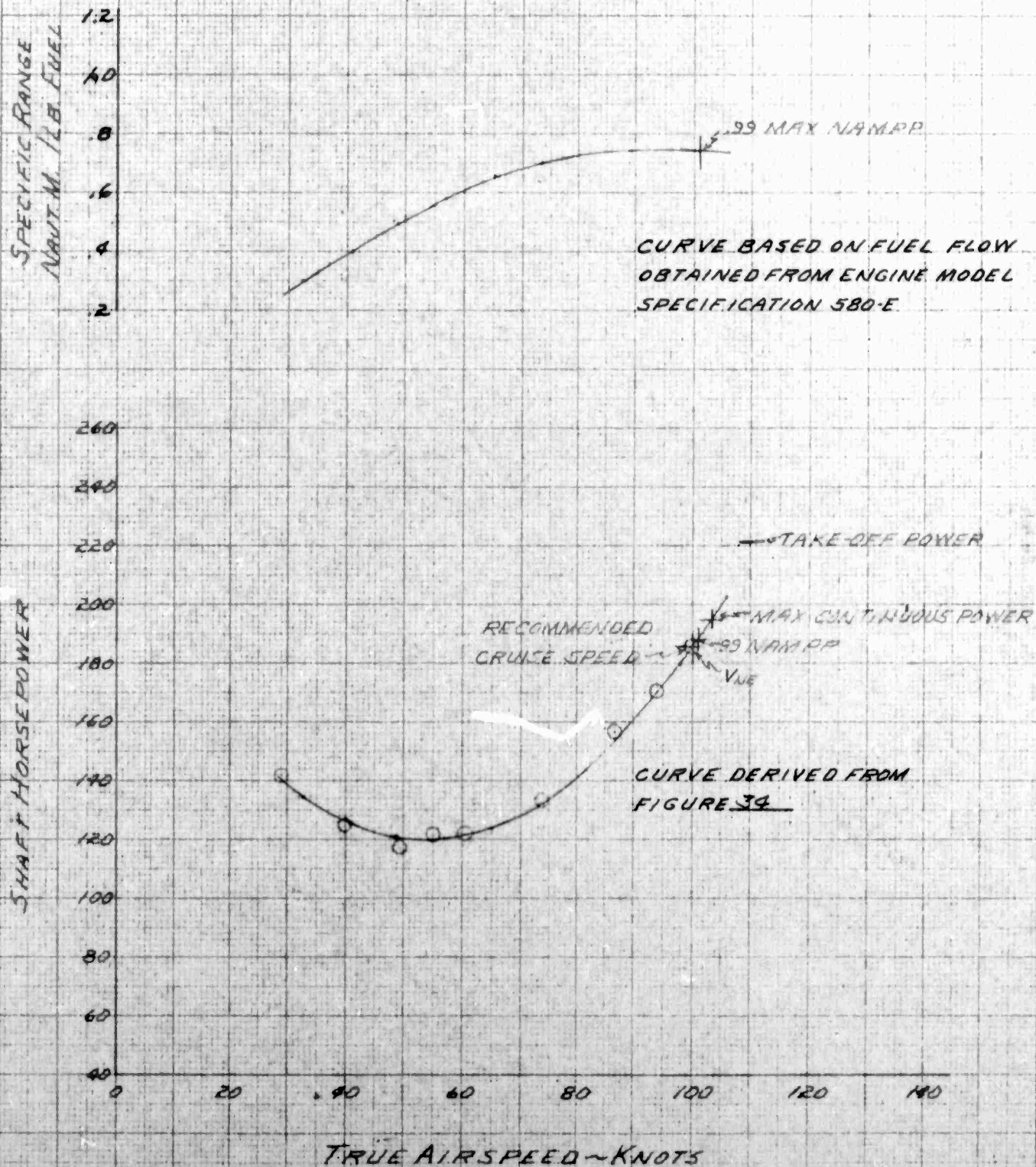


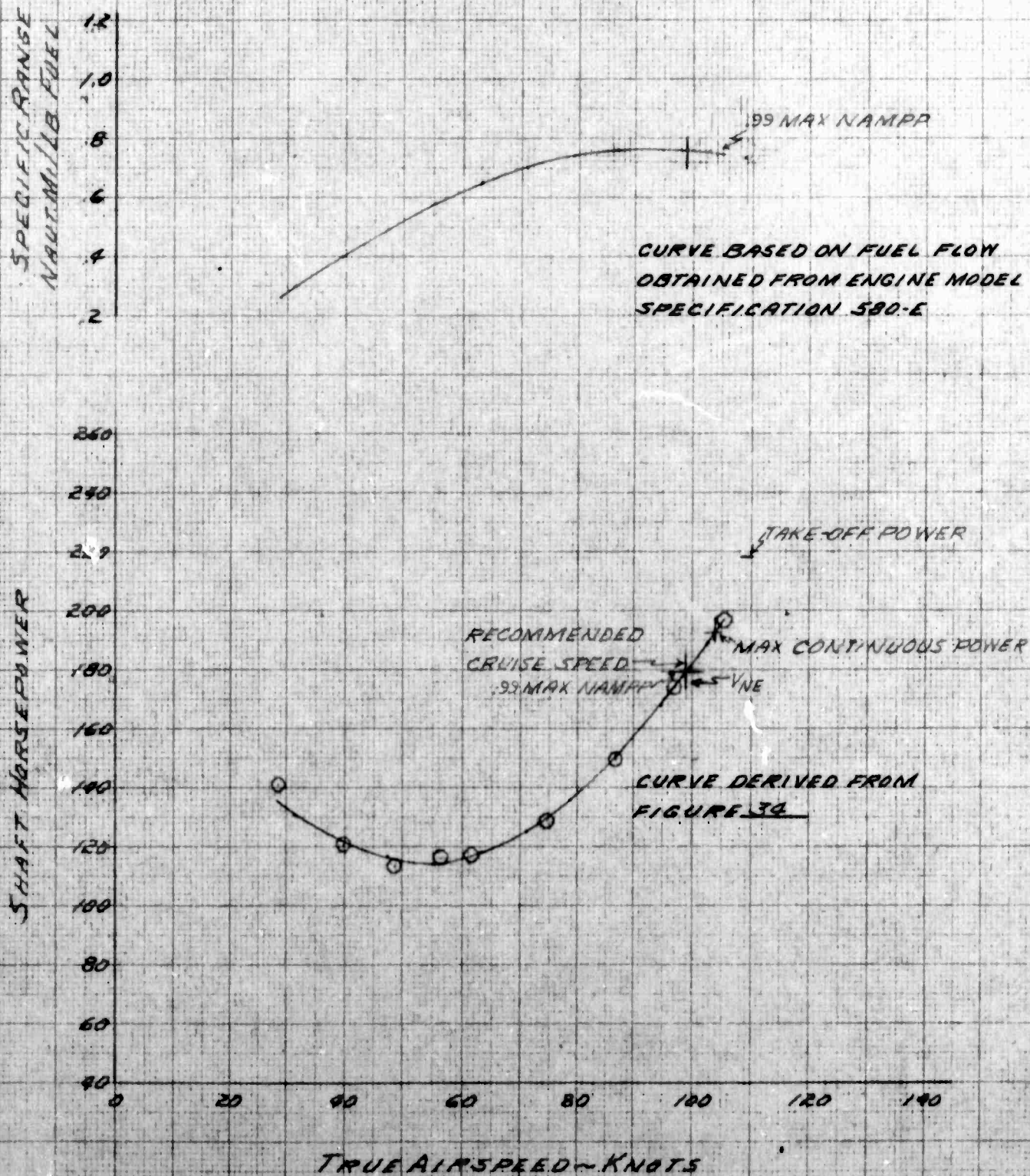
FIGURE NO. 44
LEVEL FLIGHT PERFORMANCE

OH-4A USA SIN 62-4202
GROSS WEIGHT 2960 LB.
DENSITY ALTITUDE 10370 FT.
ROTOR SPEED 394 RPM
C.G. STATION 102.2 (MID)
 $C_T = 34.30 \times 10^{-4}$
CLEAN CONFIGURATION



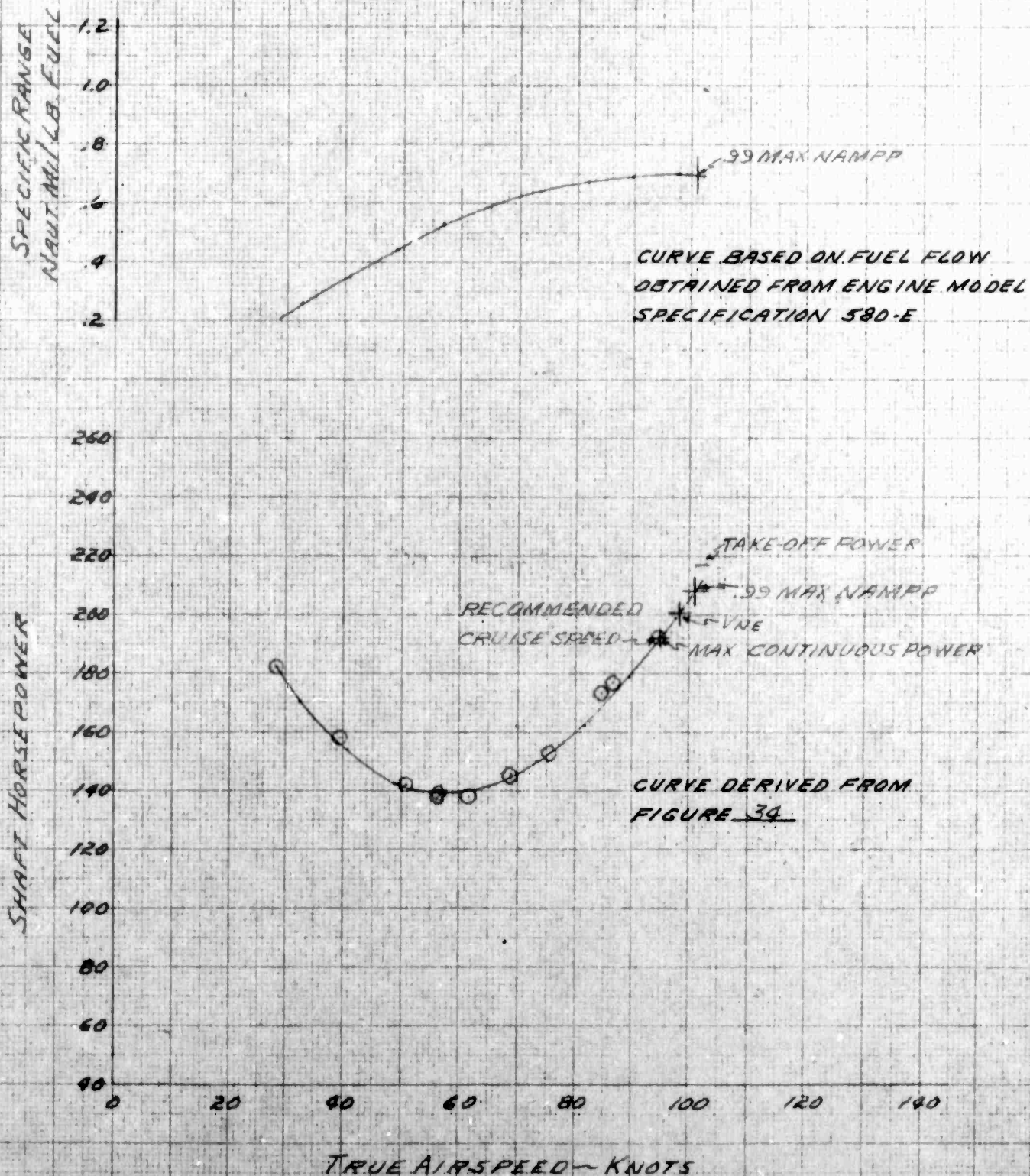
FOR OFFICIAL USE ONLY

FIGURE NO. 45
 LEVEL FLIGHT PERFORMANCE
 OH-4A USA SIN 62-4202
 GROSS WEIGHT - 2435 LB
 DENSITY ALTITUDE - 10730 FT
 ROTOR SPEED - 382 RPM
 C.G. STATION 101.2 (MID)
 $C_T = 36.48 \times 10^{-6}$
 CLEAN CONFIGURATION

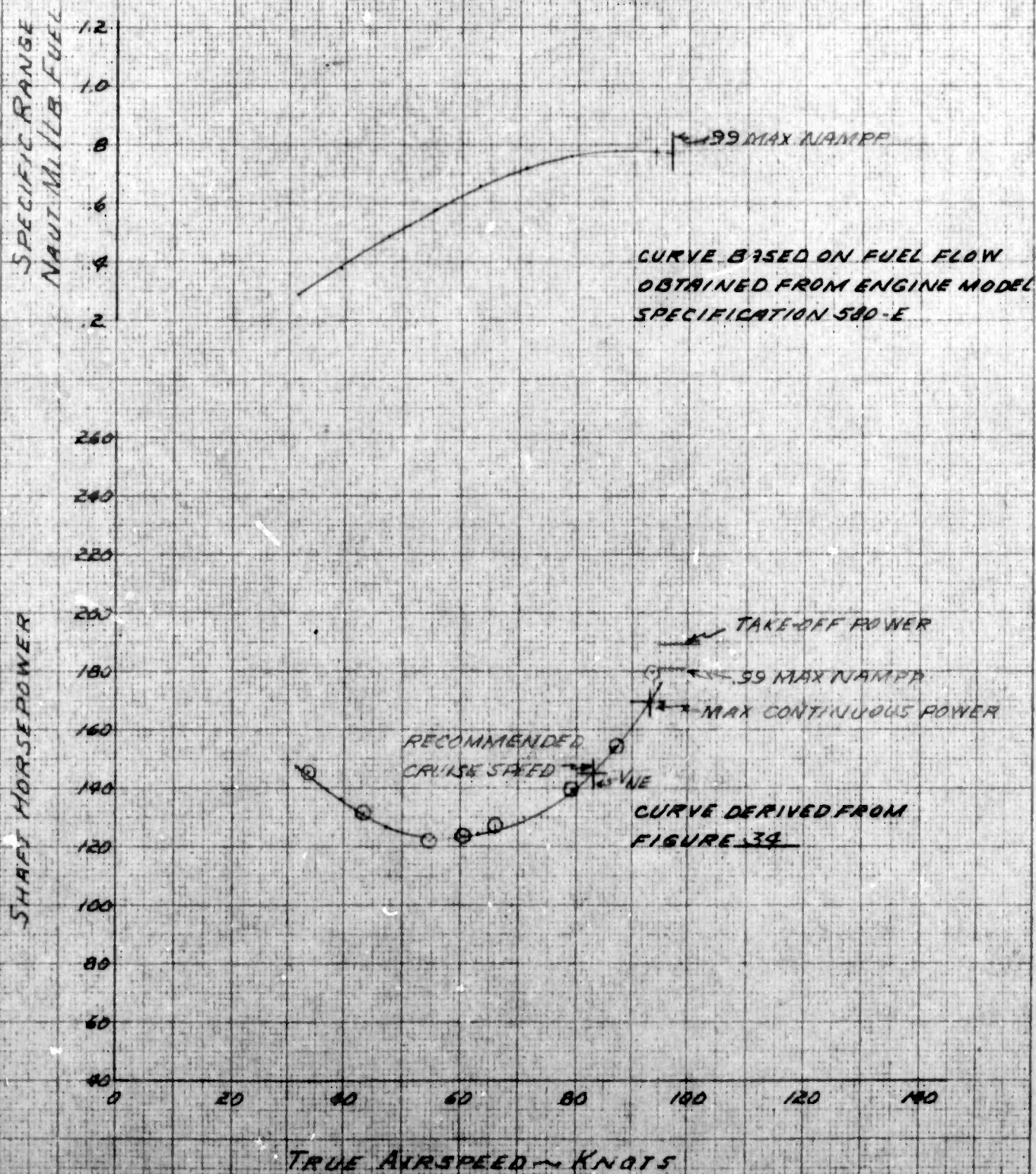


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FIGURE NO. 46
 LEVEL FLIGHT PERFORMANCE
 OH-4A USA S/N 62-4202
 GROSS WEIGHT - 2804 LB.
 DENSITY ALTITUDE - 10910 FT.
 ROTOR SPEED - 394 RPM
 C.G. STATION 101.6 (MID)
 $C_T = 39.82 \times 10^{-6}$
 CLEAN CONFIGURATION



FOR OFFICIAL USE ONLY

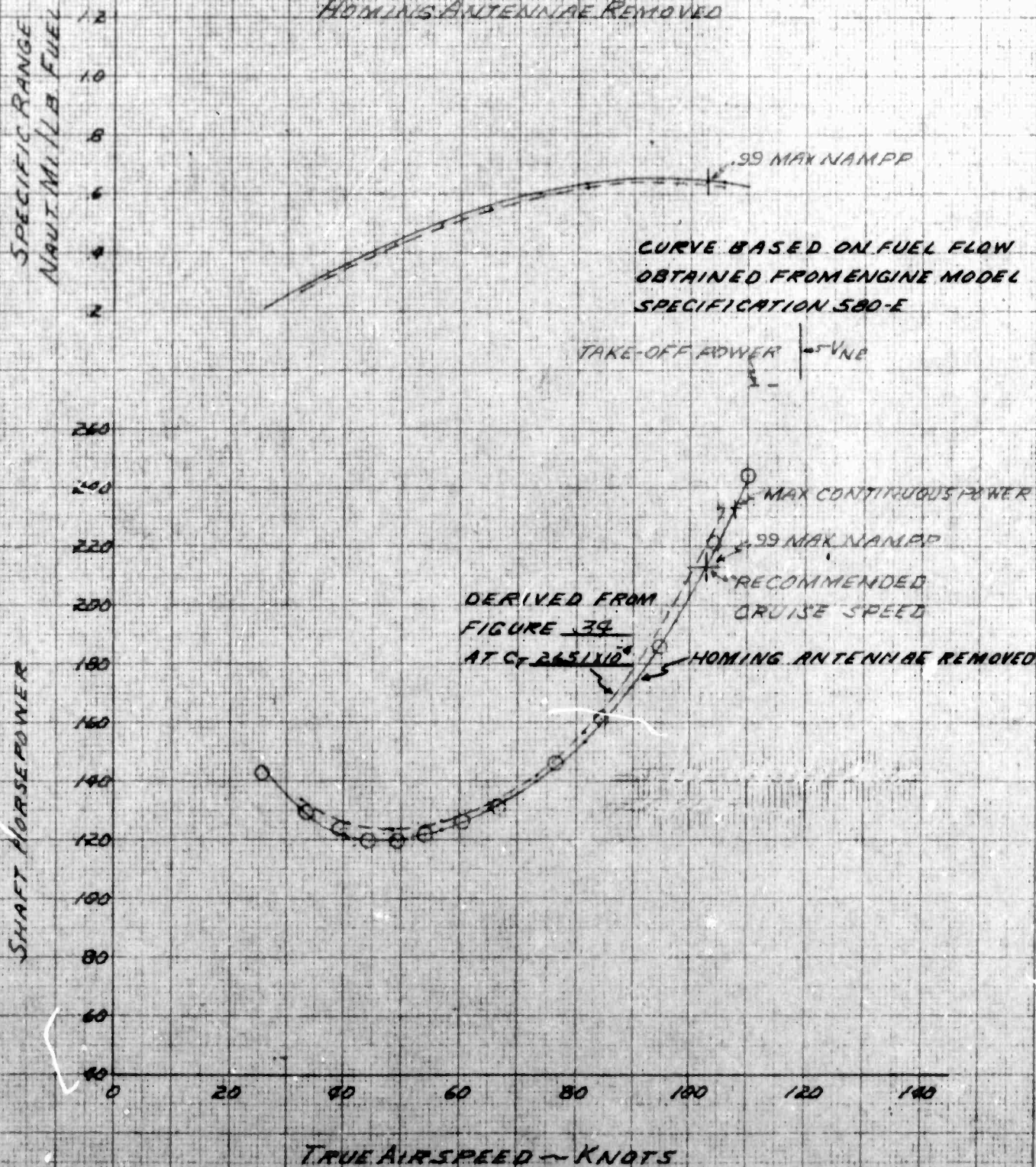
[illegible]

FOR OFFICIAL USE ONLY

FIGURE NO. 48
LEVEL FLIGHT PERFORMANCE

OH-4A USA S/N 62-4202
GROSS WEIGHT - 2435 LB
DENSITY ALTITUDE - 2225 FT
ROTOR SPEED - 394 RPM
C.G. - STATION 102.25 (MID)
 $C_T = 26.51 \times 10^{-4}$

CLEAN CONFIGURATION
HOMING ANTENNAE REMOVED



OH-9A USASIN 62-4202
GROSS WEIGHT-2397 LB
DENSITY ALTITUDE-5765 FT
ROTOR SPEED-394 RPM
C.G. STATION 102.4 (MID)
 $C_T = 2903 \times 10^{-4}$
XM-7 INSTALLED

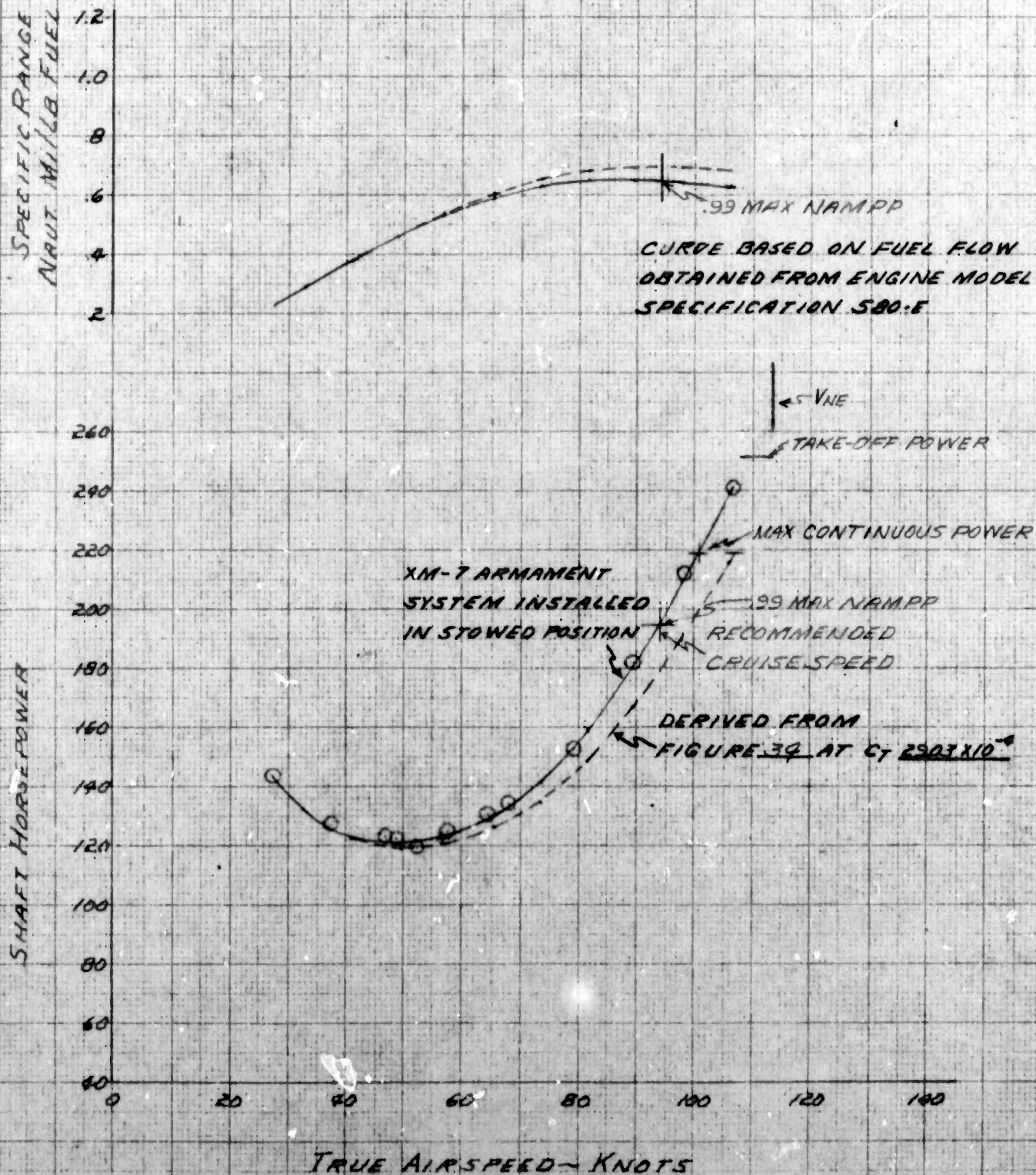


FIGURE NO 50
 LEVEL FLIGHT PERFORMANCE
 OH-4A USA S/N 62-4202
 GROSS WEIGHT 2411 LB
 DENSITY ALTITUDE 1690 FT
 ROTOR SPEED 394 RPM
 C.G. STATION 102.1 (MID)
 $C_T = 25.83 \times 10^{-4}$
 XM-8 INSTALLED

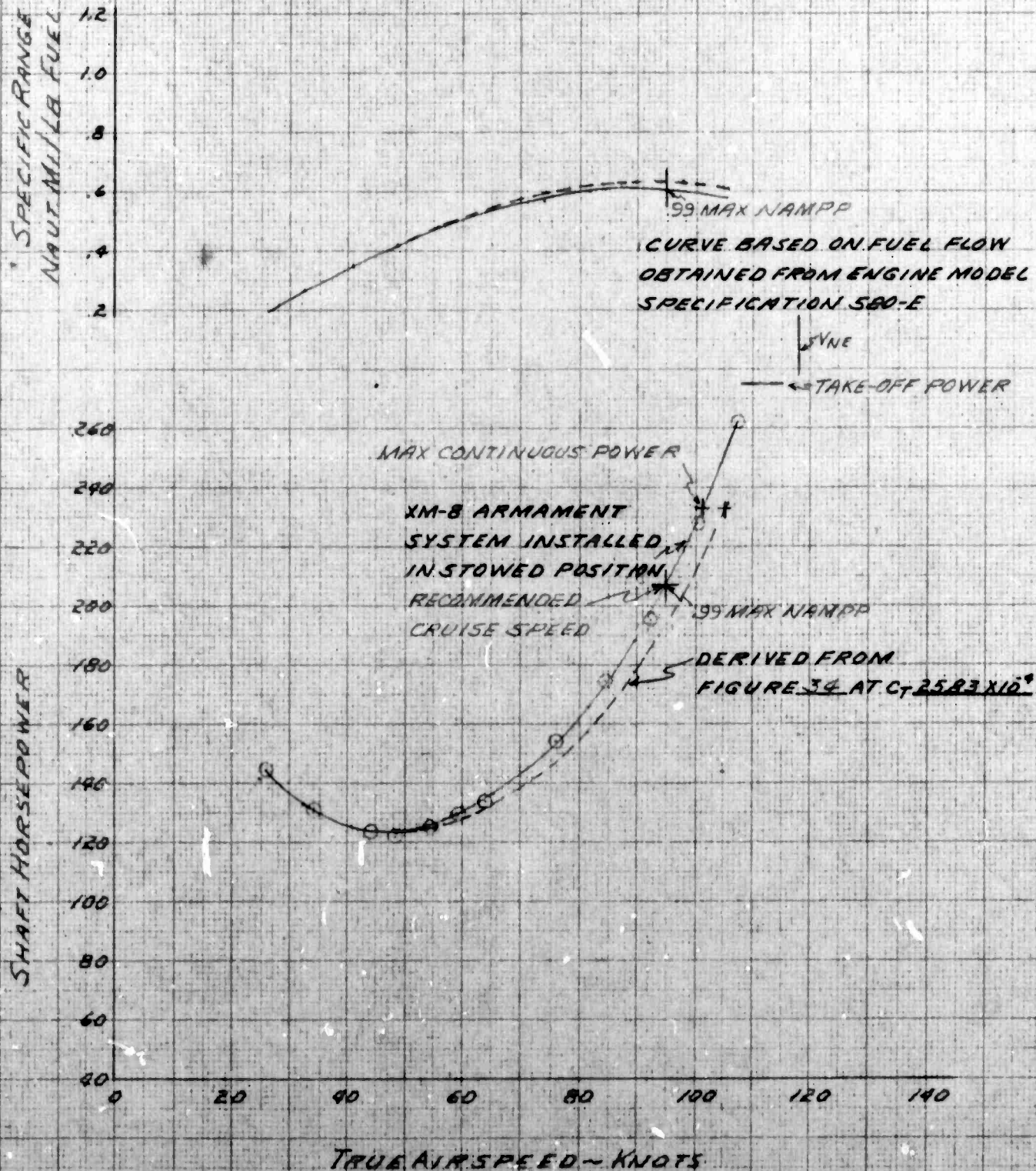


FIGURE No. 51
LEVEL FLIGHT PERFORMANCE

OH-4A USA SIN 62-4202
GROSS WEIGHT - 2387 LB.
DENSITY ALTITUDE - 590 DFT
ROTOR SPEED - 394 RPM
C.G. - STATION 102.1 (Mid)
 $C_T = 29.03 \times 10^{-4}$
XM-8 INSTALLED

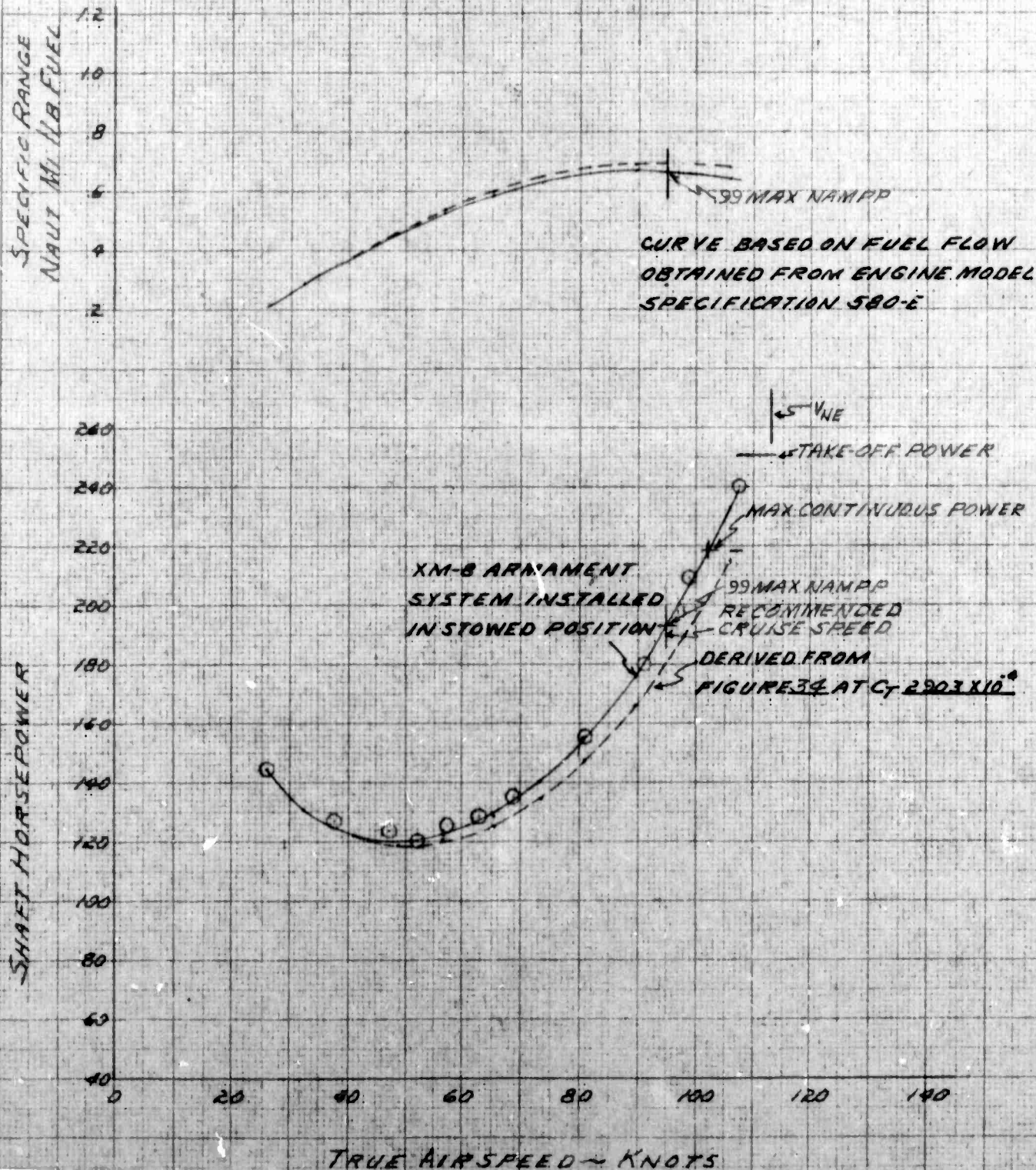
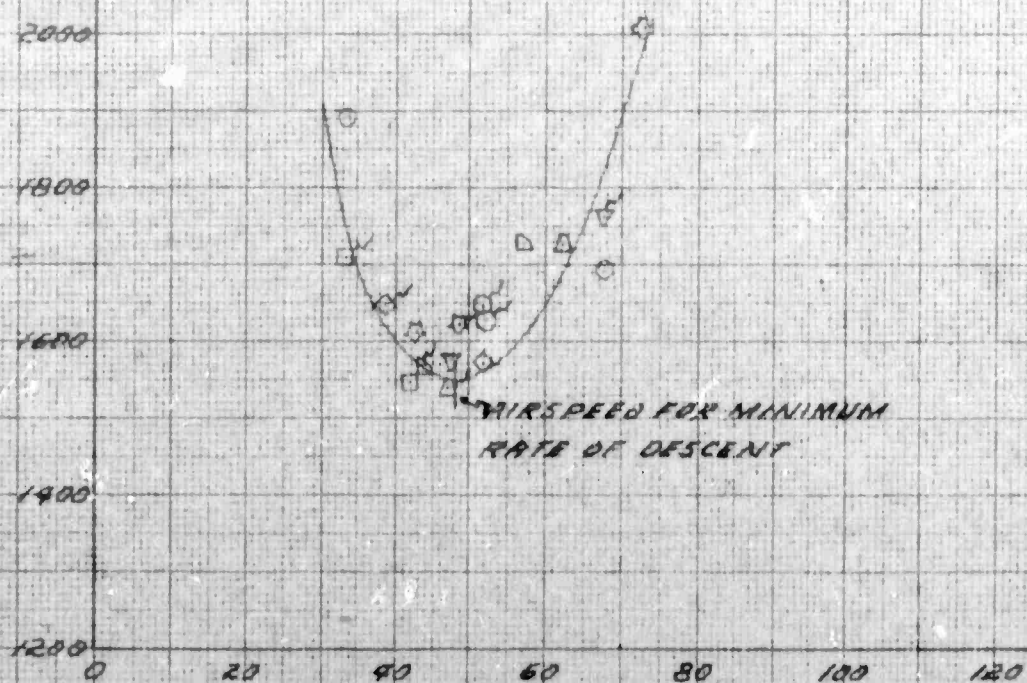


FIGURE NO 52
 AUTOROTATION DESCENTS
 OH-4A USA S/N 62-4202
 ROTOR SPEED - 394 RPM
 C.G. STATION - 101.5 (MID)
 CLEAN CONFIGURATION

SYMBOL	DENSITY ALTITUDE FT	GROSS WEIGHT LB
O	5000	2447
□	10000	2411
◇	5000	2411
D	5000	2454
D	5000	2437
○	5000	2421
○	5000	2402
○	5000	2514
○	5000	2497
○	5000	2470
○	5000	2469
○	5000	2328
○	10000	2409
○	5000	2400
○	5000	2396
○	5000	2436
○	5000	2491

RATE OF DESCENT - FT/MIN



CALIBRATED AIRSPEED - KNOTS

FIGURE NO. 53
 AUTOROTATION DESCENTS
 OH-4A USA SING 2-4202
 AIRSPEED: 98.5 KNOTS CAS
 C.G. STATION 101.5 (MID)
 CLEAN CONFIGURATION

SYMBOL	DENSITY ALTITUDE-FT	GROSS WEIGHT-LB
○	5000	2913
□	5000	2788
◇	5000	2859
▷	5000	2880
△	5000	2908
○	10000	2187
□	10000	2211
▷	10000	2298

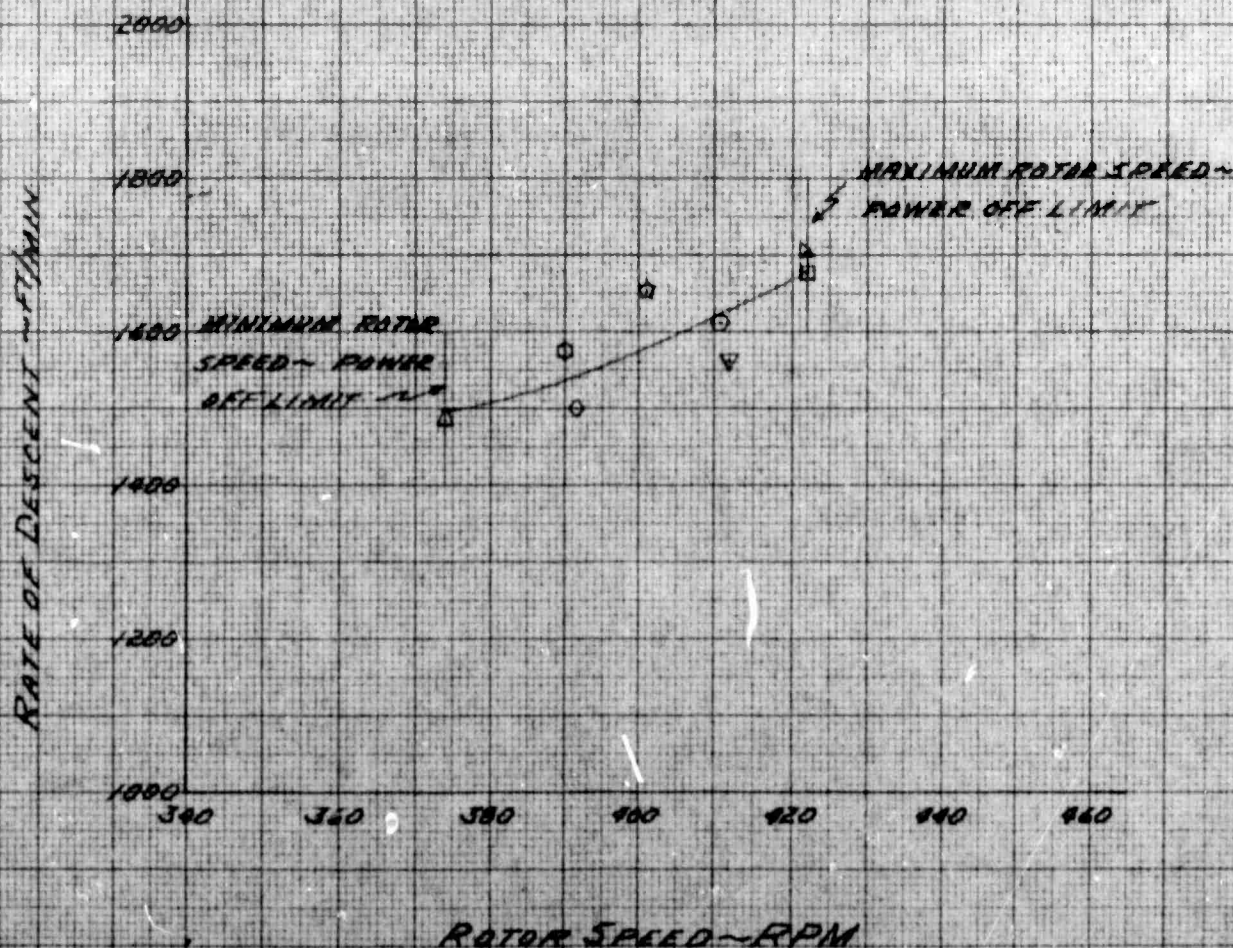


FIGURE NO 54
 AUTOROTATION DESCENTS
 OH-4A USA S/N 62-4202
 ROTOR SPEED - 399 RPM
 GROSS WEIGHT - 2438 LB
 DENSITY ALTITUDE - 5000 FT
 C.G. STATION 101.5 (MID)
 CLEAN CONFIGURATION

NAMT/1000 FEET

1.2
1.0
0.8
0.6
0.4
0.2



AIRSPEED FOR MINIMUM
 ANGLE OF DESCENT

2400

2300

CURVE DERIVED
 FROM FIGURE

2000

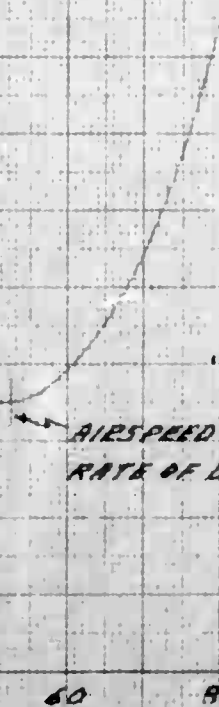
1800

1600

1400

1200

RATE OF DESCENT - FPM MIN



AIRSPEED FOR MINIMUM
 RATE OF DESCENT

20

40

60

80

100

120

TRUE AIRSPEED - KNOTS

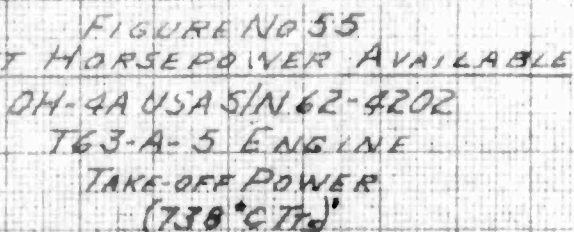
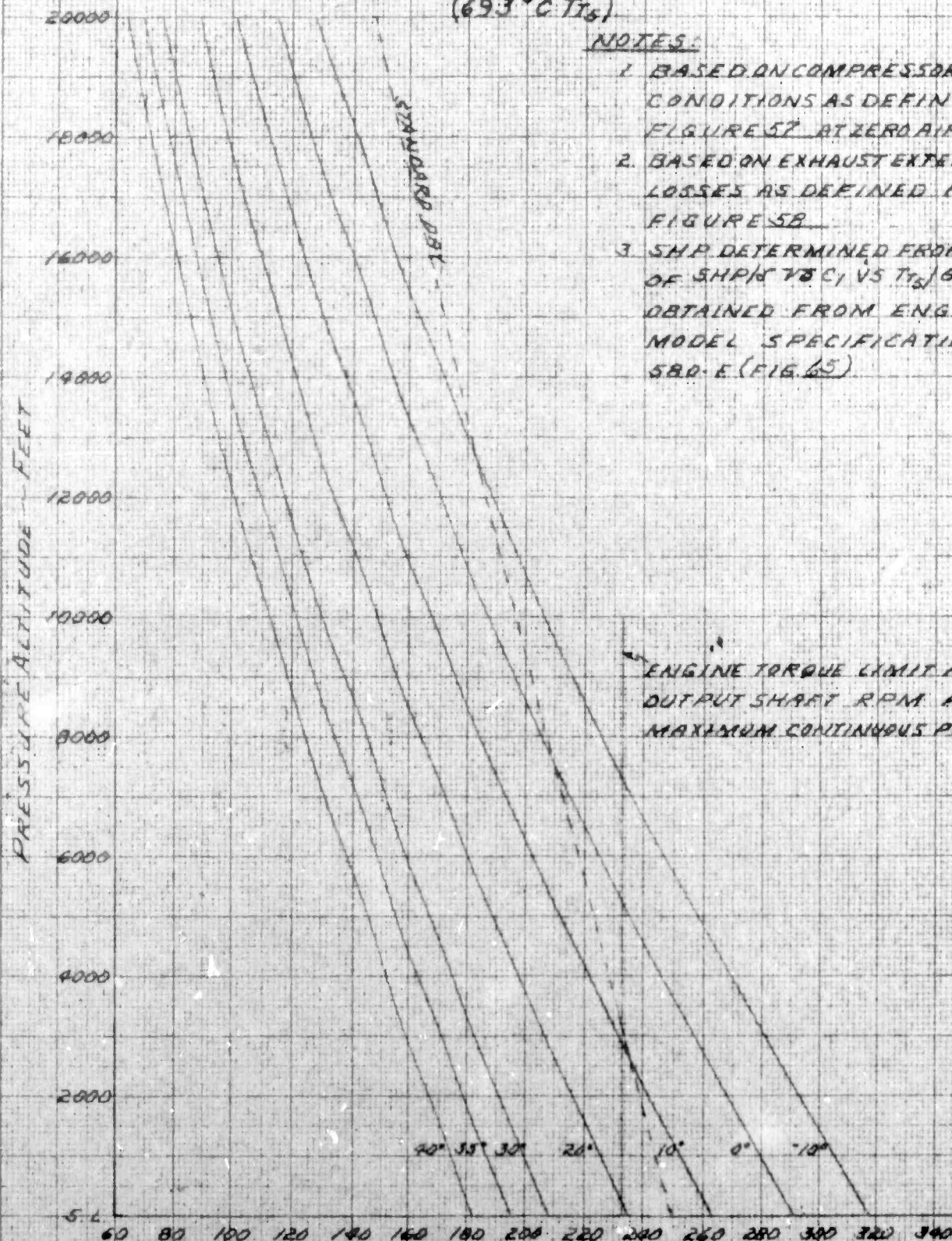


FIGURE NO. 56
SHAFT HORSEPOWER AVAILABLE
DH-4AUSA SINE2-4202
T63-A-5 ENGINE
MAXIMUM CONTINUOUS POWER
(69.3°C T_{ts})

NOTES:

1. BASED ON COMPRESSOR INLET CONDITIONS AS DEFINED IN FIGURE 57 AT ZERO AIRSPEED
2. BASED ON EXHAUST EXTENSION LOSSES AS DEFINED IN FIGURE 58
3. SHP DETERMINED FROM CURVE OF SHP/578 C₁ VS T_{ts}/80 C₂ OBTAINED FROM ENGINE MODEL SPECIFICATION 580-E (FIG. 65)

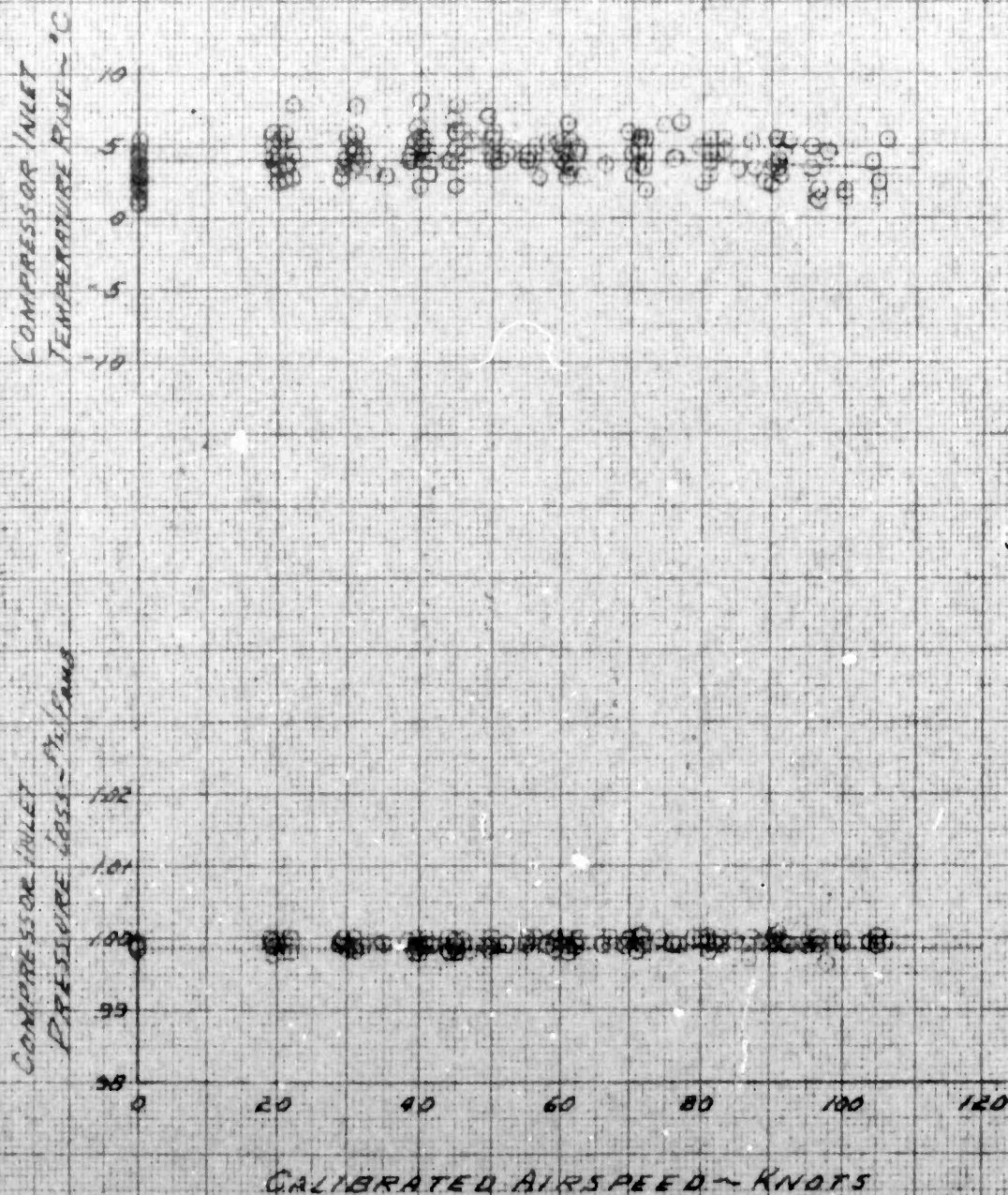


SHAFT HORSEPOWER AVAILABLE

FIGURE No. 57
ENGINE CHARACTERISTICS
OH-4A USA S/N 62-4202
T63-A-5 S/N 400040

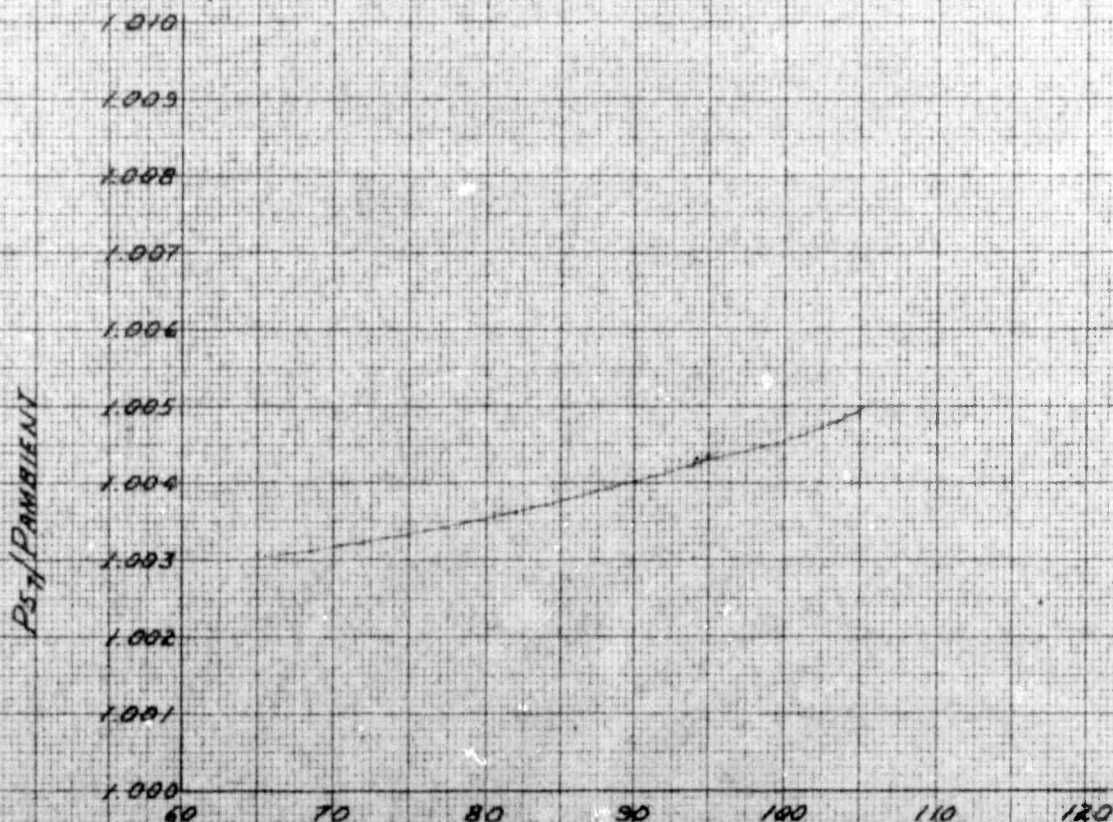
NOTES:

1. P_{t_1} IS COMPRESSOR INLET
TOTAL PRESSURE
2. POINTS WERE OBTAINED
DURING STABILIZED FLIGHT
REF. FIGURES 34 THROUGH 51.



04-9A USA S/N 62-4202
763-A-5 S/N 400040

1. BASED ON ALLISON TEST STAND DATA.
2. P_{52} - STATIC PRESSURE AT EXIT OF ENGINE EXHAUST COLLECTOR.
3. P_{AMB} - STATIC PRESSURE AT EXIT OF BELL EXHAUST EXTENSION.
4. T_0 BASED ON COMPRESSOR INLET TOTAL TEMPERATURE.
5. 100% N_2 5120 RPM.



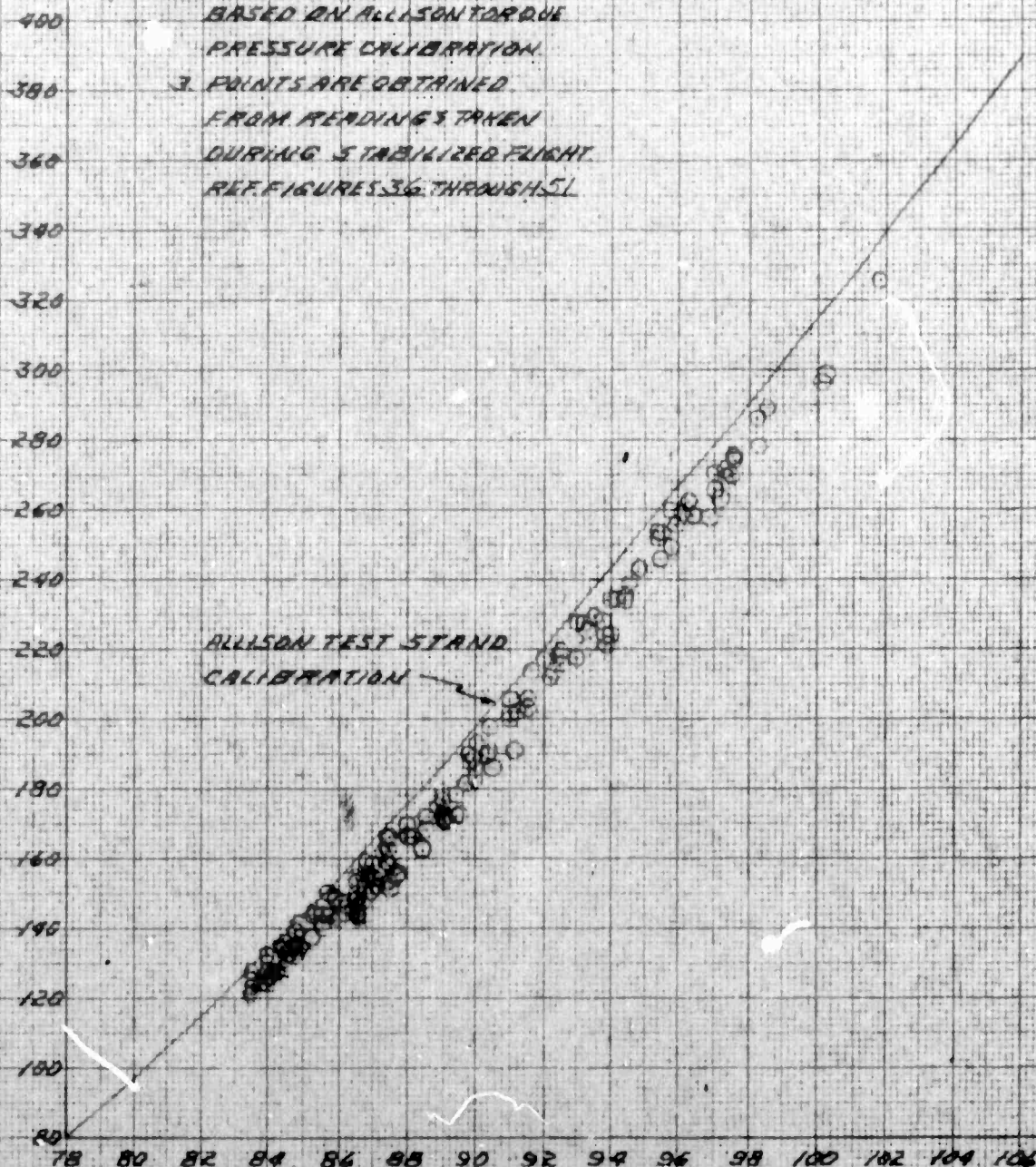
GAS PRODUCER SPEED ~ N/TS ~ % RPM

FIGURE NO. 59
ENGINE CHARACTERISTICS
OH-4A USA S/N 62-4202
T63-A-5 S/N 400040

NOTES:

1. N_1 , V_0 AND C , BASED ON COMPRESSOR INLET TOTAL PRESSURE AND TEMPERATURE.
2. SHAFT HORSEPOWER BASED ON ALLISON TORQUE PRESSURE CALIBRATION.
3. POINTS ARE OBTAINED FROM READINGS TAKEN DURING STABILIZED FLIGHT REF. FIGURES 36 THROUGH 51.

REFERRED SHAFT HORSEPOWER ~ SHP/STC,



REFERRED GAS PRODUCER SPEED ~ N_1/V_0 ~ % RPM

FIGURE No 60
ENGINE CHARACTERISTICS
OH-4A USAF/N62-4202
T63-A-5 SN 400040

NOTES:

1. J , V_0 , C , AND C_3 BASED ON
COMPRESSOR INLET
TOTAL PRESSURE AND
TEMPERATURE
2. SHAFT HORSEPOWER
BASED ON ALLISON TORQUE
PRESSURE CALIBRATION.
3. POINTS ARE OBTAINED
FROM READINGS TAKEN
DURING STABILIZED FLIGHT
REF. FIGURES 34 THROUGH 51.

REFERRED FUEL FLOW - $W_p/STOC_3$ - LB/HR.

250
240
230
220
210
200
190
180
170
160
150
140
130
120
110
100
90

ALLISON TEST STAND
CALIBRATION

80 100 120 140 160 180 200 220 240 260 280 300 320 340 360 380 400

REFERRED SHAFT HORSEPOWER - SHP/STOC₁

FIGURE NO. 61
ENGINE CHARACTERISTICS
OH-4A USAS/N 62-4202
TG3-A-5 S/N 400040

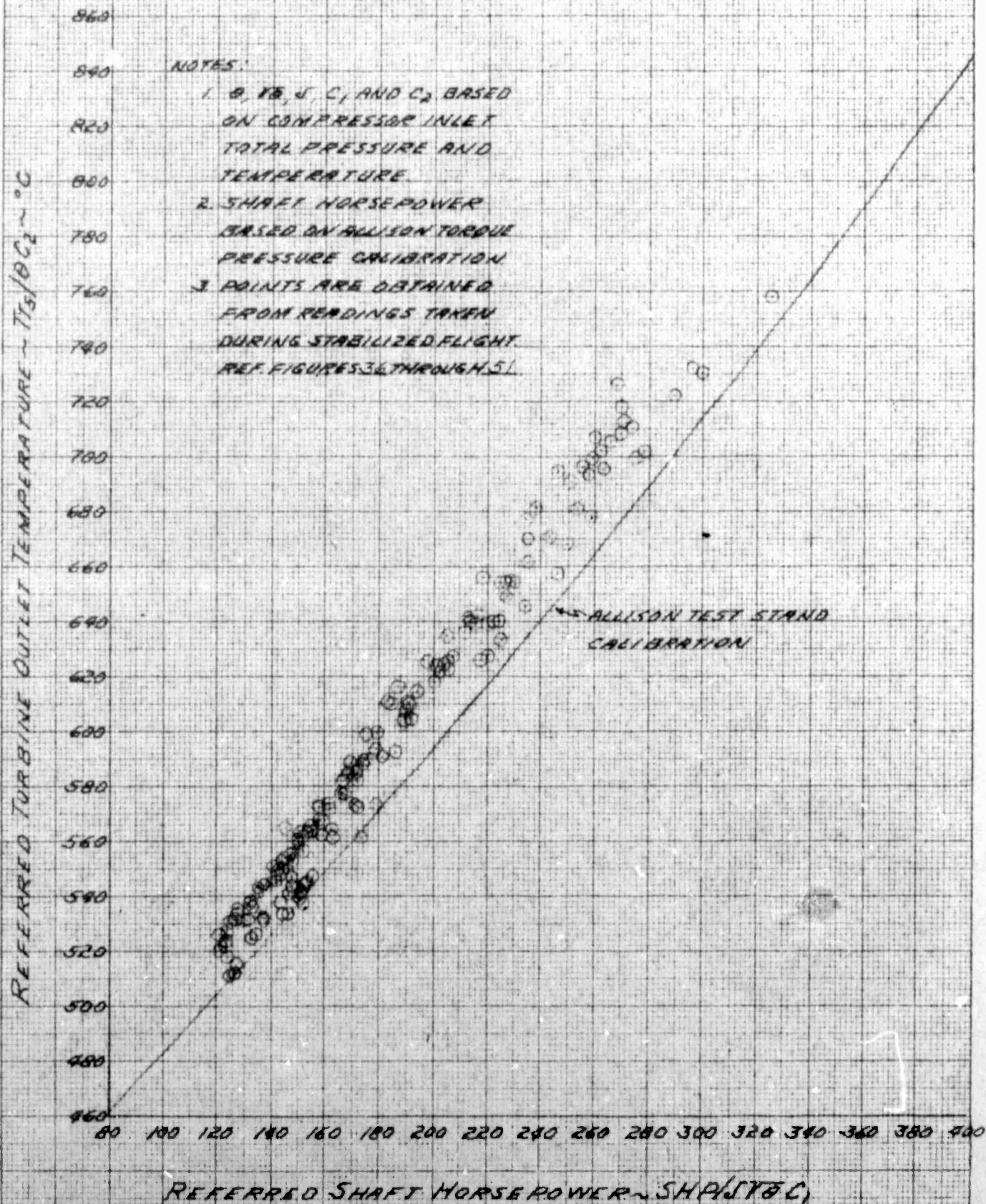


FIGURE No 62
ENGINE CHARACTERISTICS
OH-4A USA S/N 62-4202
T63-A-5 S/N 400040

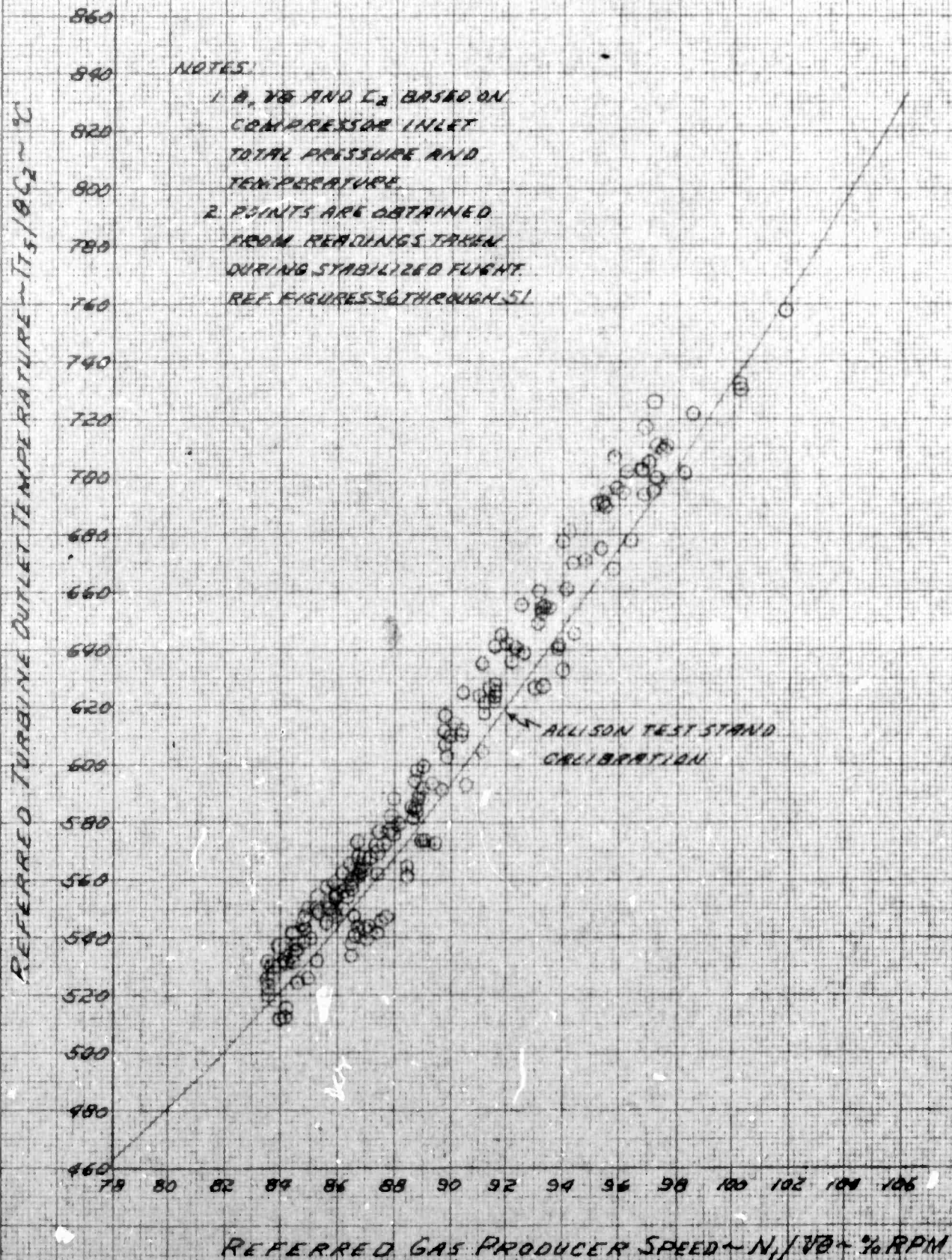
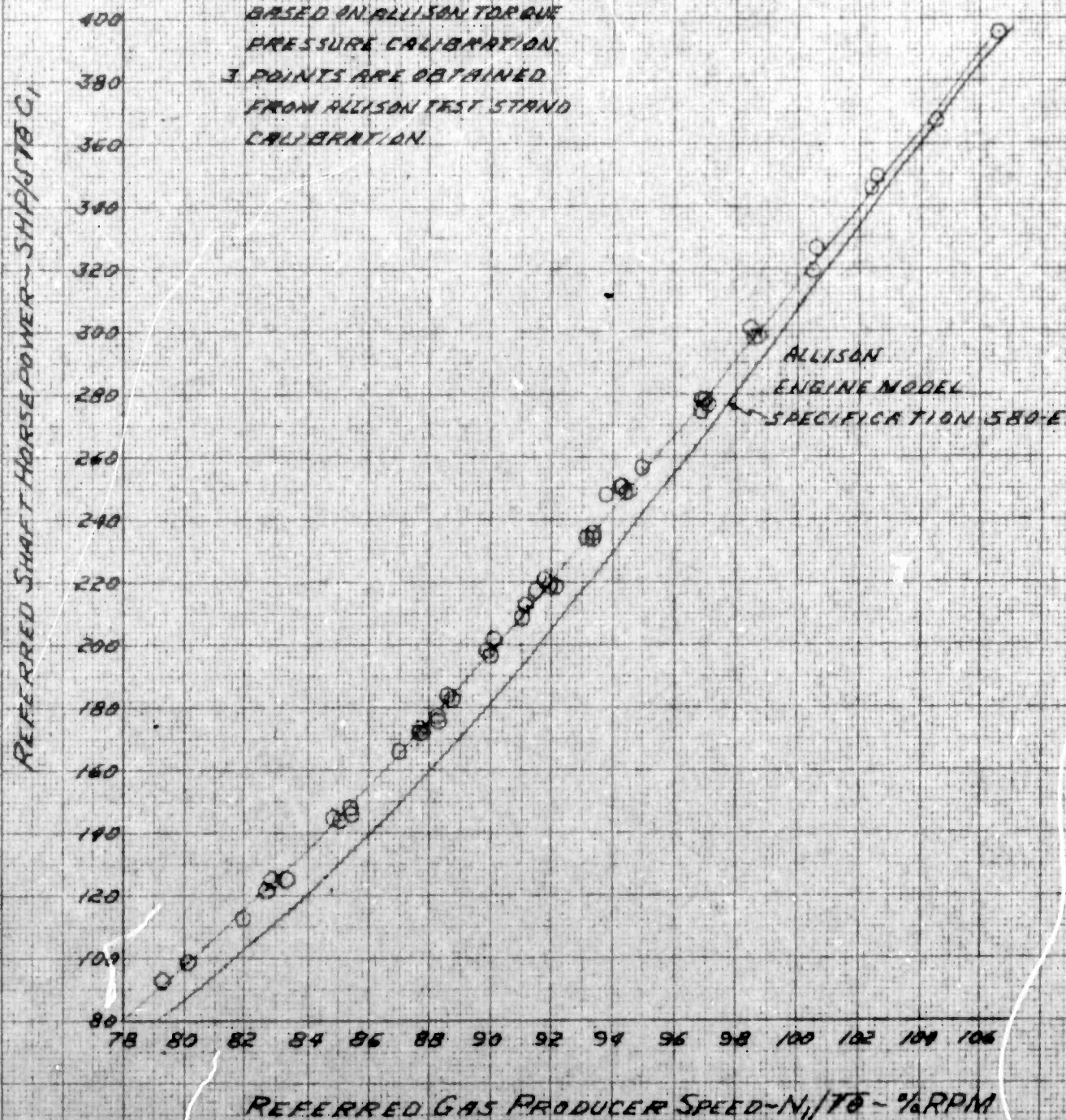


FIGURE NO 63
ENGINE CHARACTERISTICS
OH-4A USA S/N 62-4202
T63-A-5 S/N 400040

NOTES:

1. η , T_0 AND C_1 BASED ON
COMPRESSOR INLET
TOTAL PRESSURE AND
TEMPERATURE
2. SHAFT HORSEPOWER
BASED ON ALLISON TORQUE
PRESSURE CALIBRATION.
3. POINTS ARE OBTAINED
FROM ALLISON TEST STAND
CALIBRATION.



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FIGURE NO. 64
ENGINE CHARACTERISTICS
OH-4A USA S/N 62-4202
T63-A-5 S/N 400040

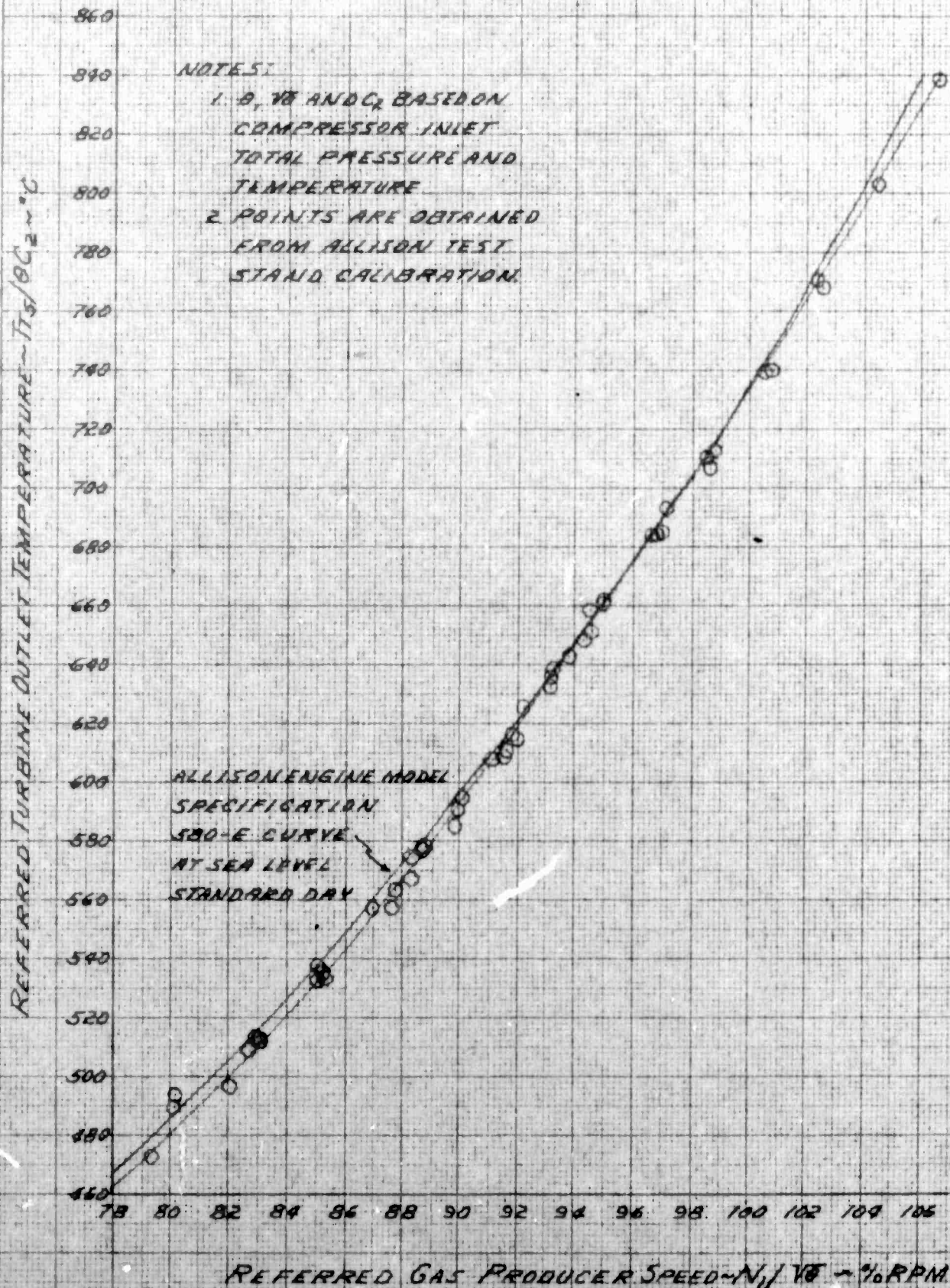
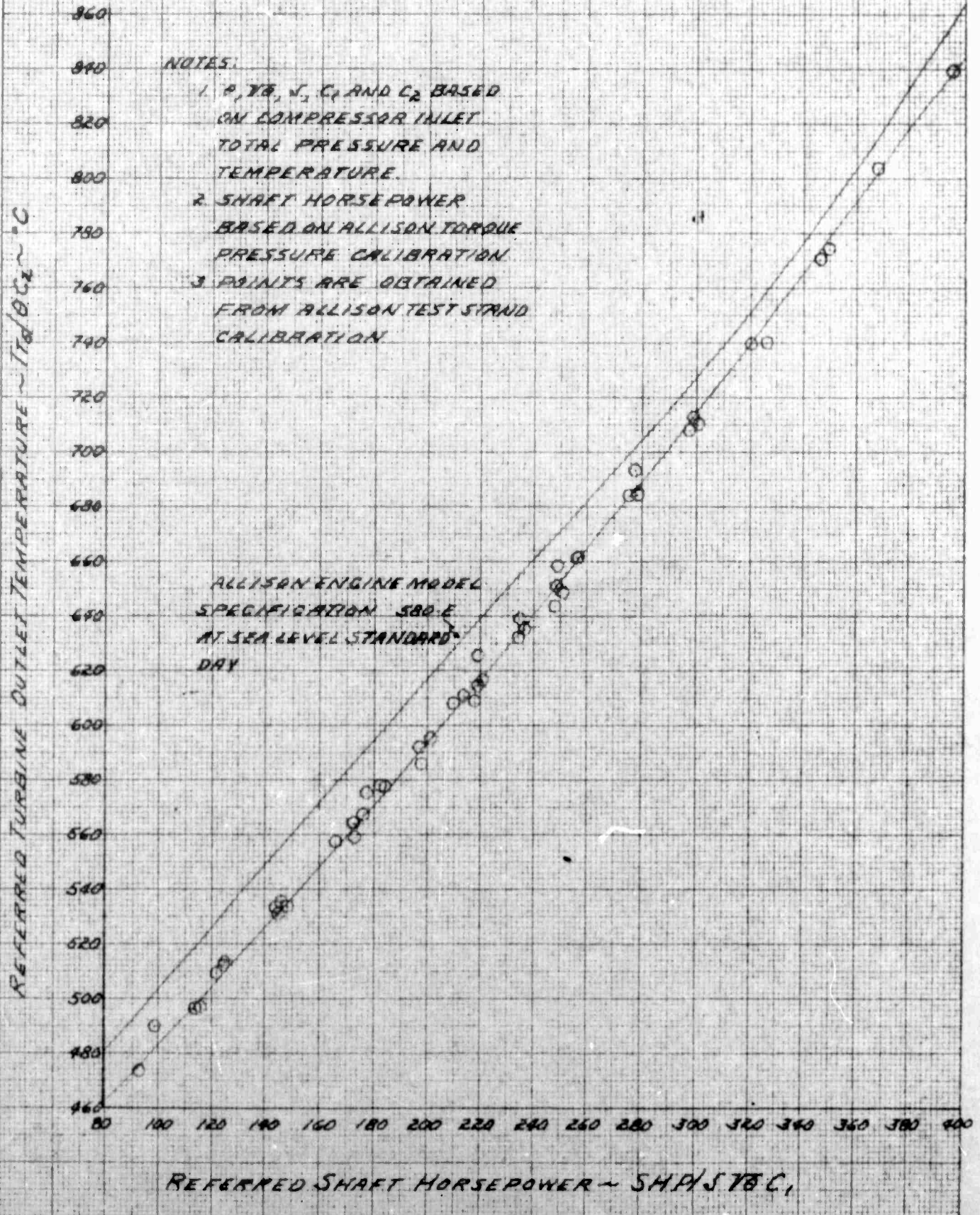


FIGURE NO. 65
ENGINE CHARACTERISTICS
OH-4A USA S/N 62-4202
T63-A-5 S/N 400040

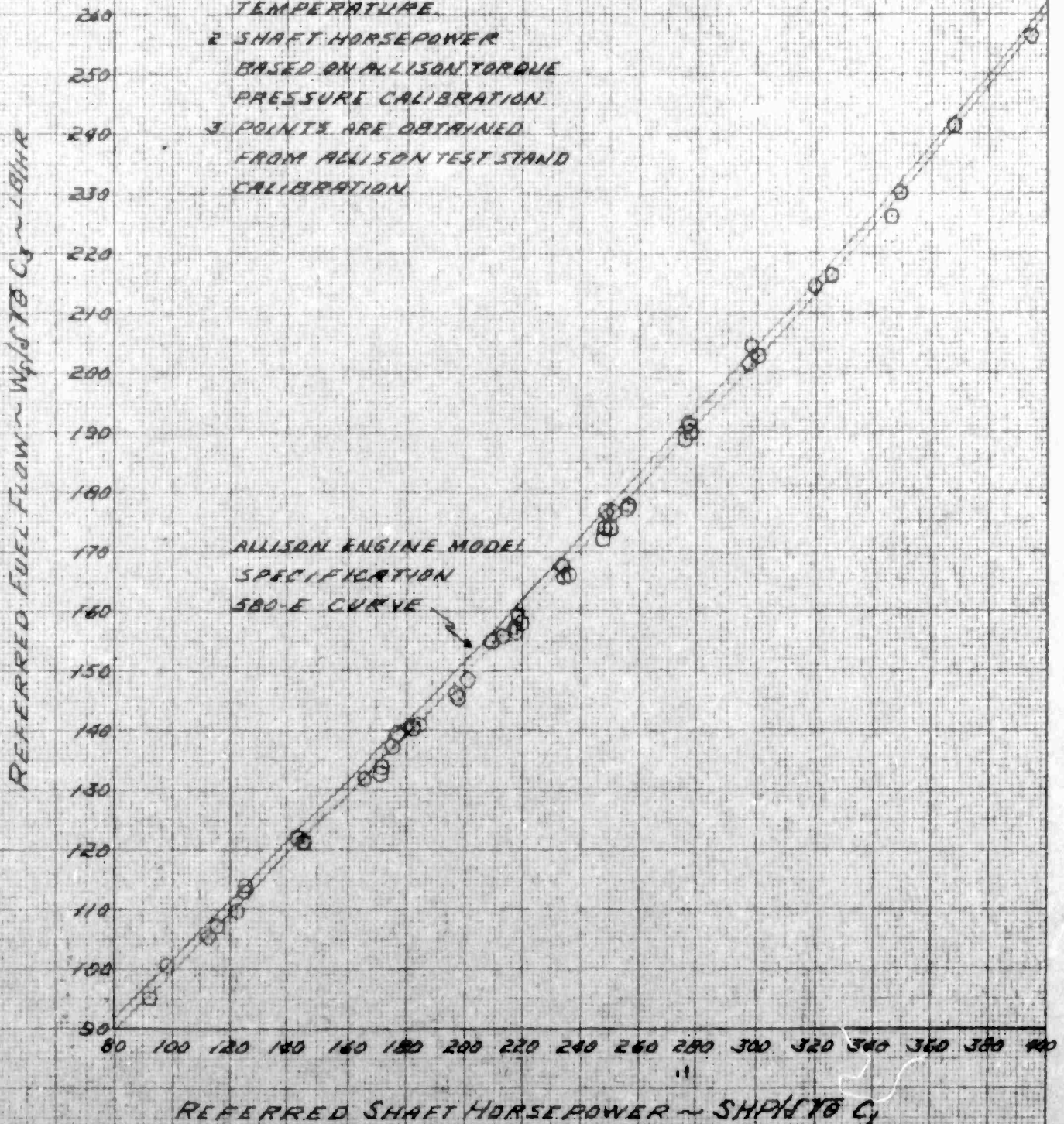


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FIGURE NO. 66
ENGINE CHARACTERISTICS
OH-4A SIN 62-4202
T63-A-5 SIN 400040

NOTES:

1. η , η_B , C_1 , AND C_2 BASED ON COMPRESSOR INLET TOTAL PRESSURE AND TEMPERATURE.
2. SHAFT HORSEPOWER BASED ON ALLISON TORQUE PRESSURE CALIBRATION.
3. POINTS ARE OBTAINED FROM ALLISON TEST STAND CALIBRATION.



[illegible][illegible]

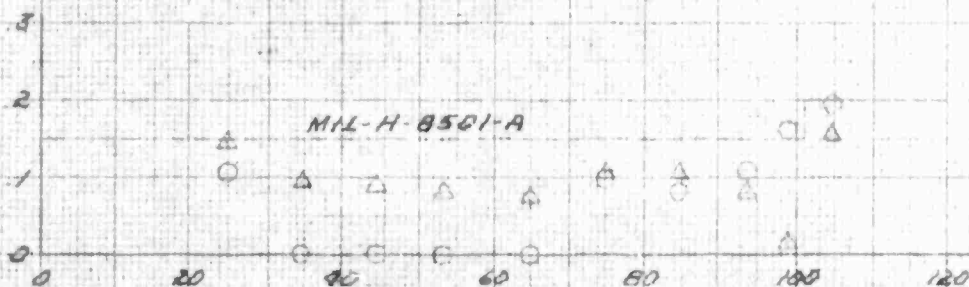
2

FIGURE NO. 68
VIBRATION CHARACTERISTICS
OH-4A USA S/N 62-4202

ROTOR RPM = 394
GROSS WEIGHT = 2431 LB
DENSITY ALTITUDE = 827 FT
C.G. = STATION 102.15 (MID)

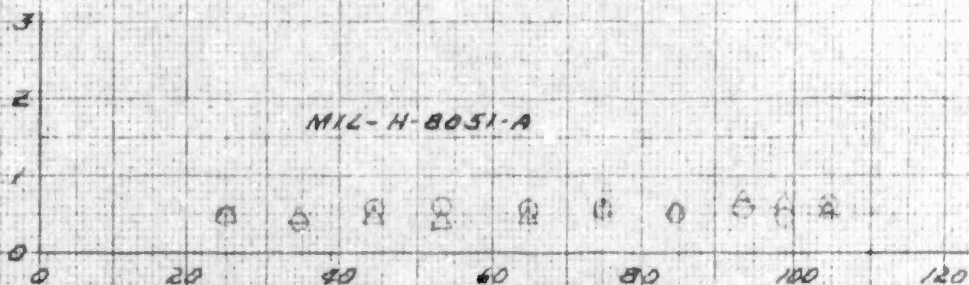
○ PILOT LATERAL
△ PILOT VERTICAL

4/REV (F. 2628 CPS)



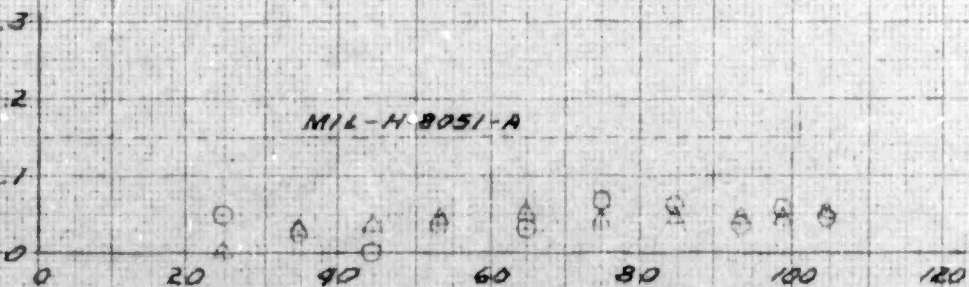
TRUE AIRSPEED ~ KNOTS

2/REV (F. 1314 CPS)



TRUE AIRSPEED ~ KNOTS

1/REV (F. 657 CPS)



TRUE AIRSPEED ~ KNOTS

SINGLE AMPLITUDE
VIBRATION ACCELERATION - G'S

APPENDIX II

GENERAL AIRCRAFT INFORMATION

Aircraft Dimensions, Design Data, FAA Type Inspection Authorization Limitations, Weight and Balance, Instrumentation, Engine Model Specification 580-E Inaccuracies and Engine Operation

1. Sources of Information

See Part I, Section 3, Appendix II.

2. Description of Aircraft and Systems

2.1 Aircraft Design Data

See Part I, Section 3, Appendix II.

2.2 Aircraft Systems

2.2.1 Electrical System

See Part I, Section 3, Appendix II.

2.2.2 Power Plant

The T63-A-5 turbo-shaft engine has a nominal rating of 250 shaft horsepower (SHP). As installed in the OH-4A, the engine is limited by either the output shaft torque or the gas producer turbine outlet temperature (T_{t5}). For maximum continuous operation these limits were 204 pound-feet torque at 6000 rpm (233 SHP) or 693 degrees centigrade (C) T_{t5} , whichever is reached first. For takeoff power (maximum of 5 minutes continuous operation), these limits are 240 pound-feet torque (275 SHP) or 738 degrees C.

The engine is a free turbine type. The compressor consists of 6 axial stages and 1 centrifugal stage. Compressor speed at 100 percent is 51,120 rpm. The combustor section consists of a single chamber into which a regulated flow of fuel is injected to support continuous combustion. The power turbine has 2 axial stages. Power turbine speed at 100 percent is 35,000 rpm. The high speed of the power turbine is reduced in the accessory gear box to 6000 rpm for the engine output speed. Engine operated accessories are also driven from the accessory gear box.

The DP-D3 gas turbine fuel control is pneumatically operated by compressor discharge air. The fuel control senses input from 3 sources. These sources are the pilot's twist grip, the fly-ball governor connected to the gas producer, and the turbine governor.

In addition, both altitude compensation and temperature compensation are provided. The function of the fuel control is to integrate the inputs so that the power turbine speed selected by the pilot is maintained under varying load demands.

A steady-state "droop" is built into the fuel control. This means that when the engine load is increased or decreased, the speed of the power turbine will change slightly. The droop is required to avoid rotor-engine dynamic instability and "hunting" during steady-state operation. In an attempt to eliminate some of the undesirable effects of the droop, a droop compensator cam is installed in the linkage between the collective pitch control and the power turbine governor. This cam converts collective control movement to an input to the power turbine governor which anticipates the changing engine load. The cam was designed to reduce transient droop and to eliminate steady-state droop during rapid collective pitch applications. Operation of this droop compensator, as installed, was marginally satisfactory throughout the test program.

There is no provision for emergency control in the case of a fuel control failure. In the case of an uncontrolled overspeed in the power turbine, rotor speed and engine power may be controlled through pilot coordination of collective pitch control and twist grip rotation as in an ungoverned reciprocating engine.

2.2.3 Landing Gear

See Part I, Section 3, Appendix II.

2.2.4 Fuel System

See Part I, Section 3, Appendix II.

2.2.5 Control System

See Part I, Section 3, Appendix II.

3.0 TIA Limitations

See Part I, Section 3, Appendix II.

4.0 Weight and Balance

The test aircraft was weighed prior to installation of test instrumentation. The weighing was done in a closed hangar using an electronic weighing kit. As weighed, the aircraft gross weight was 1574 pounds with the longitudinal C.G. located at Station 108.4.

In order to provide a basis for verifying whether the manufacturer's helicopter met the performance guarantees, the empty weight was defined to be the weight of OH-4A, USA S/N 62-4202, as delivered to the U. S. Army Aviation Test Activity, with corrections for the increase in weight of government furnished equipment, and corrections for items installed but not part of the contract empty weight (See Section 1, paragraph 1.1.0). With these corrections, the empty weight used was 1596 pounds (full oil and trapped fuel). Oil capacity was 9-1/2 pounds.

The manufacturer's performance guarantees were at a specified gross weight. This weight is defined as:

Normal Gross Weight = Empty weight + useful load

Empty Weight = 1596 pounds

Useful Load = Pilot - 200 pounds

Cargo - 400 pounds

Fuel necessary to meet normal gross weight specified in OH-4A Model Specification, paragraph 3.2.1 - 377 pounds

OH-4A Model Specification, paragraph 3.2.1 states that the normal gross weight (performance guarantee weight) shall not exceed 2450 pounds \pm 5 percent; therefore, helicopter performance was calculated at 2573 pounds (2450 + 5 percent).

Weights of some items not included in the empty weight, which may be required for various missions are as follows:

Copilot flight controls - 8.1 pounds

Anti-collision light - 4.0 pounds

Ground handling wheels - 32 pounds

XM-7 armament system - 375 pounds
(full ammunition)

XM-8 armament system - 325 pounds
(full ammunition)

After installation of test instrumentation, the helicopter was again weighed. The basic weight (full oil and trapped fuel) was 1867 pounds with the longitudinal C.G. location at Station 105.95.

5.0 Test Instrumentation

The test instrumentation used during this evaluation was supplied, installed and maintained by the Logistics Division of the U. S. Army Aviation Test Activity. Calibration of the instrumentation was accomplished by the Instrumentation Branch of the Air Force Flight Test Center; California Division of Lockheed Aircraft Corporation; Douglas Aircraft Missile and Space Division; and the Logistics Division of the U. S. Army Aviation Test Activity.

A swivel mounted pitot-static airspeed head was installed on a nose boom mounted approximately 5 feet forward of the nose of the helicopter. The static pressure ports of this pitot-static head were the pressure source for the sensitive altimeter as well as the sensitive boom airspeed indicator. The airspeed position error for this installation is shown in Figure No. 67, Section 3, Appendix I. Sensitive instrumentation was installed prior to initiation of the test flights to measure the following parameters:

Pilot-Engineer Panel:

- Boom System Airspeed
- Standard System Airspeed
- Boom Altitude
- Rate of Climb
- Angle of Sideslip
- Free Air Temperature
- Rotor Speed
- Gas Producer Speed (N_1)
- Torquemeter Oil Pressure
- Turbine Outlet Temperature (T_{t5})
- Compressor Inlet Total Temperature
- Compressor Inlet Total Pressure
- Exhaust Gas Static Pressure
- Cockpit Absolute Pressure
- Total Fuel Used
- Photo Panel Frame Counter
- Oscillograph Record Counter

Photo Panel:

Boom Altitude
Time of Day
Free Air Temperature
Gas Producer Speed (N_1)
Torquemeter Oil Pressure
Compressor Inlet Total Pressure
Compressor Discharge Total Pressure
Combustion Static Pressure
Compressor Discharge Temperature
Fuel Control Inlet Fuel Temperature
Total Fuel Used
Photo Panel Frame Counter
Oscillograph Record Counter

Recording Oscillograph:

Gas Producer Speed
Rotor Speed
Power Turbine Speed
Gas Producer Control Lever Position
C.G. Vertical Acceleration (vibration)
C.G. Lateral Acceleration (vibration)
Pilot's Station Vertical Acceleration (vibration)
Pilot's Station Lateral Acceleration (vibration)
Pilot's Event
Engineer's Event
Bridge-Balance Voltage

6. Engine Model Specification 580-E Inaccuracies

During the course of the performance tests, inaccuracies were encountered while working with the T63-A-5 Engine Model Specification

580-E. The following information is presented to clarify this situation.

The T63-A-5 Engine Model Specification 580-E contains engine performance data that do not represent an actual engine. These data are based primarily on an uninstalled power available of 206 shaft horsepower (SHP) at 6000 feet pressure altitude, 95 degree Fahrenheit (F) ambient temperature and a sea level standard day specific fuel consumption of .71 pounds of fuel used per hour per SHP at 250 SHP. The sea level condition was a guarantee point and the SHP available at 6000 feet, 95 degrees F, was a power condition contained in an earlier model specification (580-A).

In order to construct a model specification for a new engine, tests on various engine components (i.e., compressor, combustor, turbines, etc.) are conducted by the manufacturer. From these tests, the engine performance is predicted for various altitudes and ambient temperatures. As completed engines are run and more information is obtained, the predicted performance is revised.

When the engines were calibrated for the LOM flight test programs, it became apparent that the power deterioration with increasing altitude and temperature was greater than predicted. A specification engine that would just meet the performance guarantee at sea level on a standard day would not produce 206 SHP uninstalled at 6000 feet and 95 degrees F. Such an engine would only produce 199 SHP at 6000 feet, 95 degrees F.

The engine calibration test data showed that the T63-A-5 engine had a performance margin over the 580-E specification values at sea level. The existence of this sea level power margin allowed 206 SHP to be obtained at 6000 feet, 95 degrees F. The consistent existence of this margin strengthened the T63 manufacturers decision to show 206 SHP at 6000 feet, 95 degrees F in the 580-E specification.

Prior to the start of the U. S. Army Aviation Test Activity Engineering Tests on the LOM's, referred engine performance curves were obtained from the engine manufacturer. In addition, curves of various correction factors were obtained. These curves were provided to produce 580-E specification power data. A limited check between the 580-E specification and the curves provided showed apparent agreement. As engine calibration data were received from various altitude-temperature combinations, the various parameters were reduced to referred values. These data formed a single curve, as they should, which indicated that the correction factors were reasonable. Later in the test program, a more detailed check was made of the 580-E model specification. This check revealed that the 580-E specification did not contain power data that agreed with what the actual engines were producing. It also revealed areas of considerable disagreement between the 580-E specification and the manufacturer's referred curves, whereupon another correction curve was furnished. This curve was to be used only when com-

paring the model specification with actual engine performance. When this new curve was used, there was still disagreement, although not as much as previously.

The manufacturer stated that in order to provide a sufficient power margin to allow the T63-A-5 engine to be put into production and guarantee 206 SHP at 6000 feet on a 95 degree F day, the turbine outlet temperature limit of 1360 degrees F (for takeoff power) would have to be increased to 1380 degrees F.

The engine manufacturer has provided information substantiating the deterioration of power with increasing ambient temperature as presented in this report. This information also further verifies the hot day high altitude power available as presented in this report.

From the discussions with the manufacturer, it was concluded that in order to present a technically accurate and correct picture of the LOH performance, the power deterioration with increasing altitude and temperature as determined from the referred curves would be used. These curves are presented in Figures No. 63 through 66, Section 3, Appendix I and are labeled "Engine Model Specification 580-E."

7. Engine Operation

During the first 92:15 hours of operation, no detectable deterioration of engine performance occurred in the T63-A-5 engine installed in OH-4A USA S/N 62-4202. At the end of that period, the compressor section of the engine was removed to incorporate an Engineering Order. Because dirt deposits were found, the compressor was cleaned at that time. Following the cleaning, no marked improvement of performance was noted. For nearly all of the period prior to cleaning, the testing was conducted at altitudes greater than 500 feet above the ground, except for normal takeoffs and landings from hard surfaced areas.

The compressor of the engine installed in the OH-4A, USA S/N 62-4204, required cleaning four times during this evaluation so that full power output could be maintained. The compressor was cleaned at period 91:35, 132:25, 140:40 and 145:15 hours of engine operation.

Each time cleaning was required, the preceding 3 to 4 hours of operation were performed under dusty atmospheric conditions. The first cleaning was necessary after approximately 4 hours of operation under conditions in which fine particals of dust were suspended in the atmosphere to an altitude of approximately 10,000 feet, forming a "hazy" condition. The last three cleanings were required following 2 to 3 hours of continuous operation in hovering, sideward and rearward flight, in close proximity to the ground at an unprepared site.

APPENDIX III

SYMBOLS AND ABBREVIATIONS

<u>SYMBOL</u>	<u>DEFINITION</u>	<u>UNIT</u>
TAS (Vt)	True Airspeed	Knots
CAS (Vc) (Vcal)	Calibrated Airspeed	Knots
K _t (Kt)	Knots	Knots
IAS	Indicated Airspeed	Knots
Vne	Never Exceed Airspeed	Knots
Vmax	Maximum Airspeed Attainable	Knots
V _D	Maximum Permissible Dive Speed	Knots
OGE	Out of Ground Effect	-----
IGE	In Ground Effect	-----
C.G.	Center of Gravity	Inches
GW	Gross Weight	Pounds
RPM/rpm	Revolutions per Minute	-----
°C	Degrees Centigrade	Degrees
°F	Degrees Fahrenheit	Degrees
SL	Sea Level	-----
SHP	Shaft Horsepower	-----
R/D	Rate of Descent	feet per minute
R/C	Rate of Climb	feet per minute
T/C	Time to Climb	Minutes
C _p	Power Coefficient	-----
C _T	Thrust Coefficient	-----
NAMT	Nautical Air Miles Traveled	-----

NAMPP	Nautical Air Miles Per Pound of Fuel	-----
N ₁	Gas Producer Speed	Percent rpm
N ₂	Power Turbine Speed	Percent rpm
H _d	Density Altitude	Feet
H _p	Pressure Altitude	Feet
T _{t5} (TOT)	Turbine Outlet Temperature	Degrees
ρ (rho)	Air Mass Density	$\frac{\text{lb-sec}^2}{\text{ft}^4}$